Space Transfer Concepts and Analyses for Exploration Missions

Phase Four Final Report
September 1993

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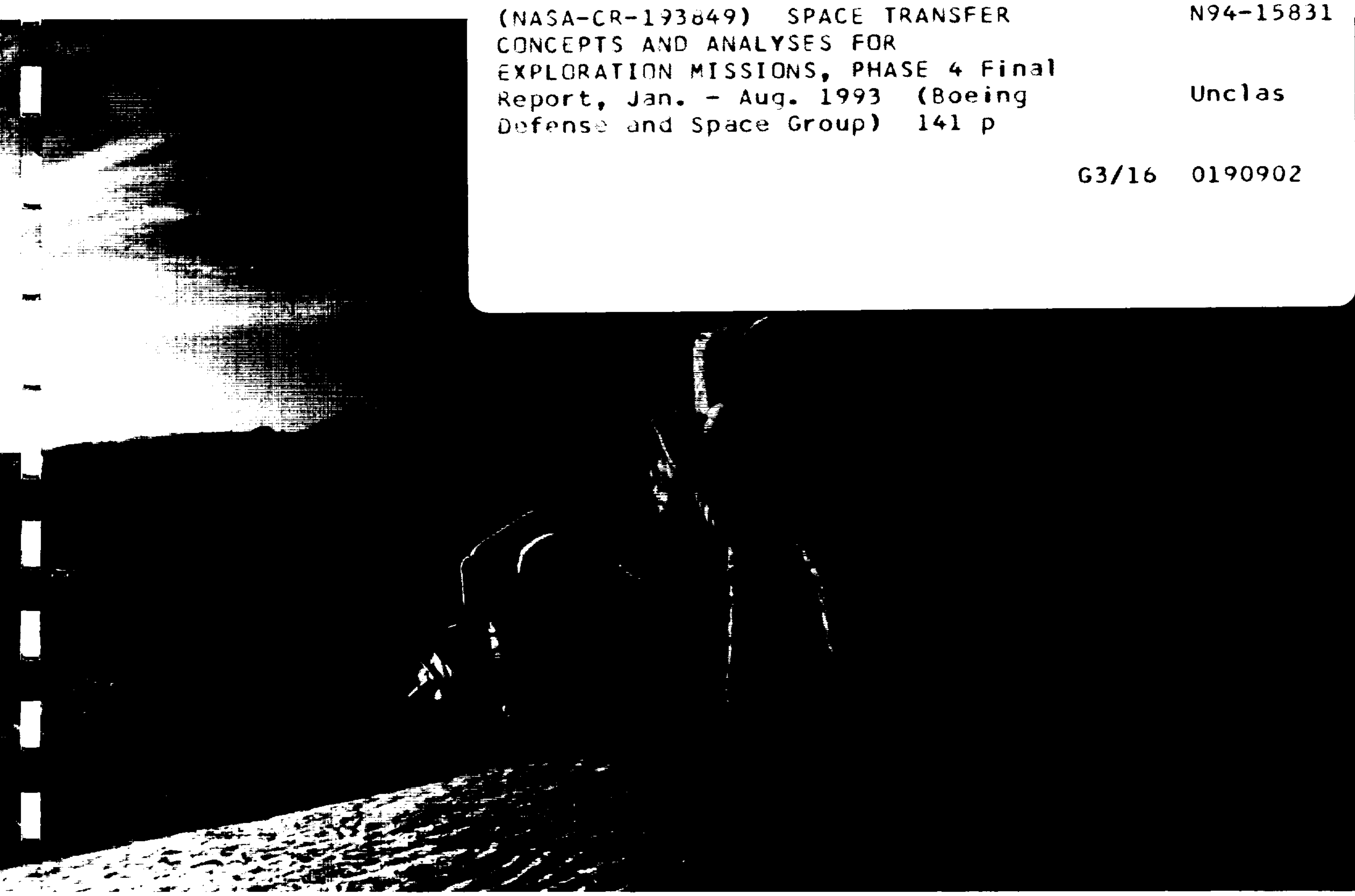

Boeing Defense and Space Group
Civil Space Product Development
Huntsville, Alabama

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Space Transfer Concepts and Analyses for Exploration Missions

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Phase 4

Final Report

September 1993

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FOREWORD

The study entitled "Space Transfer Concepts and Analyses for Exploration Missions (STCAEM)" was performed by Boeing Defense and Space Group, Huntsville, Alabama, for the George C. Marshall Space Flight Center (MSFC). The activities reported herein were carried out under Technical Directives 16, and 17 during the period January through August 1993. The Boeing program manager was Gordon Woodcock and the MSFC Contracting Officer's Technical Representative was Alan Adams. The deputy program manager at Boeing was Dr. Irwin Vas. The task activities for the studies carried out under these Technical Directives were performed by M. Appleby, P. Buddington, M. Cupples, B. Donahue, and R. Fowler.

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ABBREVIATIONS AND ACRONYMS

AB or A/B	Aerobrake
A/L	Airlock
AML	Apsidal misalignment loss
AP	All propulsive
au	Astronomical units (150 x 10^6km)
AV	Ascent Vehicle
C	(degrees) Celsius temperature
C3	Measure of trajectory energy, $(Vhp)^2$
C3L	C3 - launch
CEP	Circular error probability
C_f	Coefficient of friction
CHeCS	Crew Health Care System
CO_2	Carbon Dioxide
C_p	Specific heat at constant pressure
CRV	Crew Return Vehicle
CRYO	Cryogenic - refers to liquid oxygen and liquid hydrogen propulsion
DoD	(U. S.) Department of Defense
DoE	(U. S.) Department of Energy
DMS	Data Management System
DSM	Deep Space Maneuver
ECLSS	Environmental Control and Life Support System
EOM	End of Mission
EP	Electric Power and Distribution; also electric propulsion
ETO	Earth-to-Orbit
ETV	Earth Transfer Vehicle
EVA	Extravehicular Activity
ExPO	Exploration Program Office
FLO	First Lunar Outpost
FSD	Full Scale Development
g	Acceleration in Earth Gravities (acceleration 9.80665 m/s^2)
GN&C	Guidance Navigation and Control
hab	Habitat Module
IMLEO	Initial Mass in low Earth Orbit
IR&D	Independent Research and Development
Isp	Rocket Specific Impulse
ISRU	In-Situ Resources Utilization
IVA	Intravehicular Activity
JSC	Johnson Space Center
K	Temperature in Kelvin Units
kg	kilograms
km	kilometers
km/sec	kilometers/second
kW	kilowatts

ABBREVIATIONS AND ACRONYMS (CONTINUED)

kW_e	Kilowatts electric
kcal	kilocalories
L2	One of the Earth-Moon LaGrange (libration) points
LEO	Low Earth Orbit
LEP	Laser electric propulsion
LH_2	Liquid Hydrogen
LPLM	Lunar Pressurized Logistics Module
LSS	Life Support System
m	meters
MCC	Mid-Course Correction
MEV	Mars Excursion Vehicle
MOC	Mars Orbit Capture
MSFC	Marshall Space Flight Center
MTV	Mars Transfer Vehicle
MW	Megawatts
NASA	National Aeronautics and Space Administration
NEP	Nuclear Electric Propulsion
NERVA	Nuclear Engine for Rocket Vehicle Application
NLS	National Launch System
NTP	Nuclear Thermal Propulsion
NTR	Nuclear Thermal Rocket
NUC	Nuclear
O_2	Oxygen
ORU	Orbital Replacement Unit
p.o.	Parking orbit
psia	Pounds per square inch absolute
R&D	Research and Development
RFC	Regenerable Fuel Cell
RTG	Radioisotope thermoelectric generator
SEI	Space Exploration Initiative
SEP	Solar Electric Propulsion
Sol	1 day on Mars
SPCU	Suit Processing and Check-out Unit
SPE	Solar Proton Event
SSF	Space Station Freedom
STCAEM	Space Transfer Concepts and Analyses for Exploration Missions
STP	Standard Temperature and Pressure
t	Metric tons (1000 kg); also thickness
TBD	to be determined (unknown)
TEI	Trans-Earth Injection
TMI	Trans Mars Injection
TPS	Thermal Protection System

ABBREVIATIONS AND ACRONYMS (CONCLUDED)

V_{hp}	hyperbolic excess velocity
VSB	Venus Swingby
W	Watts
WM	Waste Management
WMF	Waste Management Facility

ABSTRACT

Earlier studies carried out under this contract covered a wide range of lunar and Mars transportation options, and lunar rove concepts and technology needs. The current report discusses the activities conducted under Technical Directives 16 and 17. Mars transportation was addressed as well as a review and update of architectures and propulsion systems.

1.0 INTRODUCTION

1.1 STUDY SCOPE

The Space Transfer Concepts and Analyses for Exploration Missions (STCAEM) study addresses in-space transportation systems for human exploration missions to the Moon and Mars. The subject matter includes orbit-to-orbit transfer vehicles, planetary landing/ascent vehicles, and the crew modules needed to form complete crew and cargo transportation systems. Also included are orbital assembly and operations facilities if these are needed for assembly, construction, recovery, storage in orbit, or processing in-space transportation systems for reuse. All propulsion and systems technologies that can be technically evaluated are open for consideration. Excluded from the study are Earth-to-orbit systems. Crew entry vehicles intended for direct Earth atmosphere entry from a lunar or planetary return trajectory are included. Capabilities of, and constraints on, Earth-to-orbit systems and their operations are parametrically considered as a boundary condition on in-space transportation systems.

1.2 REPORT SCOPE

This report represents Phase 4 of the STCAEM study. Phase 1 covered a wide range of lunar and Mars transportation options, and lunar rover concepts and technology needs. Phase 2 concentrated on Mars transportation using nuclear thermal propulsion. Phase 3 concluded certain trade studies on Mars transportation that were begun during Phase 2; most of Phase 3 was devoted to analysis of a lunar surface habitation system, the "First Lunar Outpost". Phase 4, conducted under Technical Directives 16 and 17, returned to the subject of Mars transportation with a review and update of architectures and propulsion systems. The Statements of Work for these technical directives included the following major tasks:

a. Task 1: Architecture Assessment - Assess impacts of evolution to Mars crew rotation and resupply, in-situ propellant production, and alternate mission modes

b. Task 2: Lunar Synergism - Assess the Mars transportation system synergism with the lunar program including commonality and evolution.

c. Task 3: Advanced Transportation Systems Update - Provide and update to NTP, NEP, Cryo-Aerobraking, and Mars lander concepts.

d. Task 4: Technology Assessment - Provide assessment of the technology requirements for Mars transportation system - timing, priorities and program plan outlines.

1.3 REPORT ORGANIZATION

This report is organized in three volumes. The present volume covers architecture assessment, lunar synergism, and technology assessment. In addition, new results on electric propulsion mission profile analysis are included. Review of electric propulsion systems architecture indicated that not enough significant new work had been done to merit a separate volume on electric propulsion.

The second volume is a re-issue of the Nuclear Thermal Propulsion Implementation Plan and Element Description Document: the STCAEM contract requires one of these documents be reissued whenever enough new data are available to make the prior issue obsolete. Similarly, the third volume is a re-issue that covers aerobraking and Mars landers.

2.0 ARCHITECTURE ASSESSMENT

The architecture assessment task covered certain specific issues addressed under section 2.4 below, but also addressed "understanding the implications of various mission architectures, transportation vehicle selections, mission-enhancing technologies, and mission modes", and provided our "analysis of the 'big picture' aspects of Mars missions". (Quotes from the Statement of Work)

2.1 MARS ARCHITECTURE OVERVIEW

NASA planning for human exploration missions to the Moon and Mars was resumed in NASA following publication of the Report of the President's Commission on Space, *Pioneering the Space Frontier* (the "Paine Commission Report") in 1986. This began with a study by NASA and the Los Alamos National Laboratory, *Manned Mars Missions*, which researched and updated earlier literature on the subject. The NASA-Los Alamos study actually took place during the Commission deliberations and the detailed report was released in June of 1986, a month after the Commission report.

More than a decade earlier, numerous studies of human planetary missions were conducted by NASA. These studies, from about 1963 to about 1973, investigated many mission profiles including Venus and Mars flyby missions and Mars landing, and concentrated on a Mars landing mission using nuclear thermal propulsion on an opposition mission profile. The stay time at Mars was typically described as 30 days; the missions tended to be what today would be called "flags and footprints", although a few studies included permanent bases. (Note that one 30-day stay on Mars would be nearly three times the total Apollo stay time on the Moon.) Various landing dates from 1975 to 2000 were considered.

In the mid-1960s U. S. space planners anticipated that successful conclusion of the Apollo lunar landing program would lead to establishment of modest lunar bases, or an early manned Mars landing mission or both. By about 1970 it became clear that national funding for a Mars mission would not be forthcoming in the foreseeable future, and that any extension of Apollo missions to the Moon would be confined to use of modified Apollo systems. The nation was considering decisions which would lead to the Space Shuttle program. The rationale being developed for the shuttle focused on "economic space transportation for practical applications". Support for continued study of Mars missions withered and the last ongoing studies were completed by 1973. The U. S. entered a period of more than a decade of no funding of any consequence for exploration mission studies.

In 1968, a concept for collecting solar power in space and transmitting it to Earth was published. A modest NASA-funded study of solar power satellites took place in 1972-73. In 1974, Gerard O'Neill published the results of a student design project on space colonies in Physics Today. By early 1975, these ideas had merged to the extent that a conference was held at Princeton University; the mission of space colonies would be to build solar power satellites, for energy supply to Earth, using lunar materials.

These ideas gained a certain popularity. Societies of space enthusiasts formed to promote the colonization of space. In 1984, when the Paine Commission held public hearings on the U. S. space program, space enthusiasts testified in large numbers, advocating a more aggressive space program including colonization. Dr. Paine himself was a long-time enthusiast for the exploration and eventual settlement of Mars. The Paine Commission Report called for an aggressive U. S. space program emphasizing exploration and technology advancement.

In response, NASA initiated new systems studies and planning activities for exploration missions.

2.1.1 "Case Studies"

The study effort began with a series of "case studies", each intended to develop a particular mission profile and concept or answer a specific set of questions. Two mission profile design innovations occurred early in the effort: a "split-sprint" Mars mission and two kinds of cycler (multiple-encounter) Earth-Mars trajectories.

The split sprint was as an answer to the challenge of performing a round-trip Mars mission in one year. The opposition profiles of the earlier studies normally required at least 15 months. The split sprint profile entails separate early delivery of the propellant for return to Earth from Mars, on a low-energy trajectory, and parking the propellant tanker in Mars orbit where it is used to refuel the crew mission upon Mars arrival. This means that the crew mission does not carry its return propellant on its high-energy trajectory to Mars. The advantage in initial mass in Earth orbit (IMLEO) can be greater than 2:1 for high energy, "fast" profiles.

Cycler trajectories were devised to satisfy the idea of placing an Earth-Mars-Earth transfer habitat system into a repeating transfer orbit so that it could be used on successive missions without further propulsion. Small "taxi" vehicles would be used on Earth and Mars encounters to transfer crews. Mars cargoes would be separately delivered on one-way trajectories.

A major issue addressed by the case studies was whether or not the first human Mars mission could be reduced in cost by visiting one of Mars' moons, Phobos, instead of landing. In addition, the potential benefit of extraction of propellant from Phobos, for the return trip, was considered.

Phobos' orbit about Mars is inefficient from a flight mechanics point of view. For this reason, while significant savings compared to landing were found for a mission to Phobos, the mission was still a major interplanetary expedition without the excitement of landing on the planet itself. Further, the benefits of extracting propellant from Phobos were largely negated by the added performance required, and dependence on Phobos-derived propellant on a first mission was seen as a major risk. Consequently, the idea of a first mission to Phobos was abandoned.

Continuing "case studies" focused on Mars landing missions, using cryogenic propulsion for Earth and Mars departure and aerobraking for Mars and Earth capture. Trajectories were optimized, and vehicle concepts developed, for several opportunities

2.1.2 90-Day Study

The "90-Day Study" of exploration missions followed from a speech by President Bush on July 20, 1969, establishing the Space Exploration Initiative (SEI) as NASA's next major human space mission after Space Station Freedom. This activity also marked the beginning of current Boeing involvement in exploration studies by way of the present study. Boeing supported the 90-Day Study by analyzing mission profiles, developing Mars transportation systems concepts, and performing systems trades.

The "90-Day Study" baseline scenario established a permanent base for 4 people on the Moon and had a series of missions to Mars beginning in 2015 with a Mars mission for 4 to 6 people every Mars opportunity. The space transportation concepts were "conventional", employing cryogenic propulsion and aerobraking. Figure 2-1 illustrates the STCAEM concept for a cryogenic/aerobraking vehicle developed during the "90 Day Study". Lunar missions were of course aerobraked only on Earth return. Mars missions used aerobraking for Mars and Earth arrival. Several Earth launch vehicle options were portrayed, generally based on the proposed National Launch System (NLS) or derivatives thereof. The overall program, and the projected costs, tended to be driven by space transportation. While life cycle cost estimates were not published in the report, the results were common knowledge, and the total life cycle figure was reported to be about $500 billion in 1990 dollars. While this is a large sum, the "90-Day Study" program was not derived through cost trades; it was an ambitious program; and it stretched over about 35 years. The "90-Day Study" report also included variations on the baseline scenario, some of which reduced the life cycle cost by scaling back the mission activities.

Cryogenic propulsion with aerobraking was baselined for the "90-Day Study" because it was perceived as having minimum technology risk and significant payoff. Several prior studies had indicated high payoff for aerobraking and reuse of upper stages for geosynchronous orbit and lunar missions. The earlier "case studies" had emphasized aerobraking and shown that Mars mission trajectories can be tailored to use an aerobrake to the limit of its capabilities, reducing the performance demand placed on the propulsion system. At the time of the "90-Day Study", the proposed Aeroassist Flight Experiment was in development and was expected to demonstrate a low L/D large-area aerobrake with a flight in 1994.

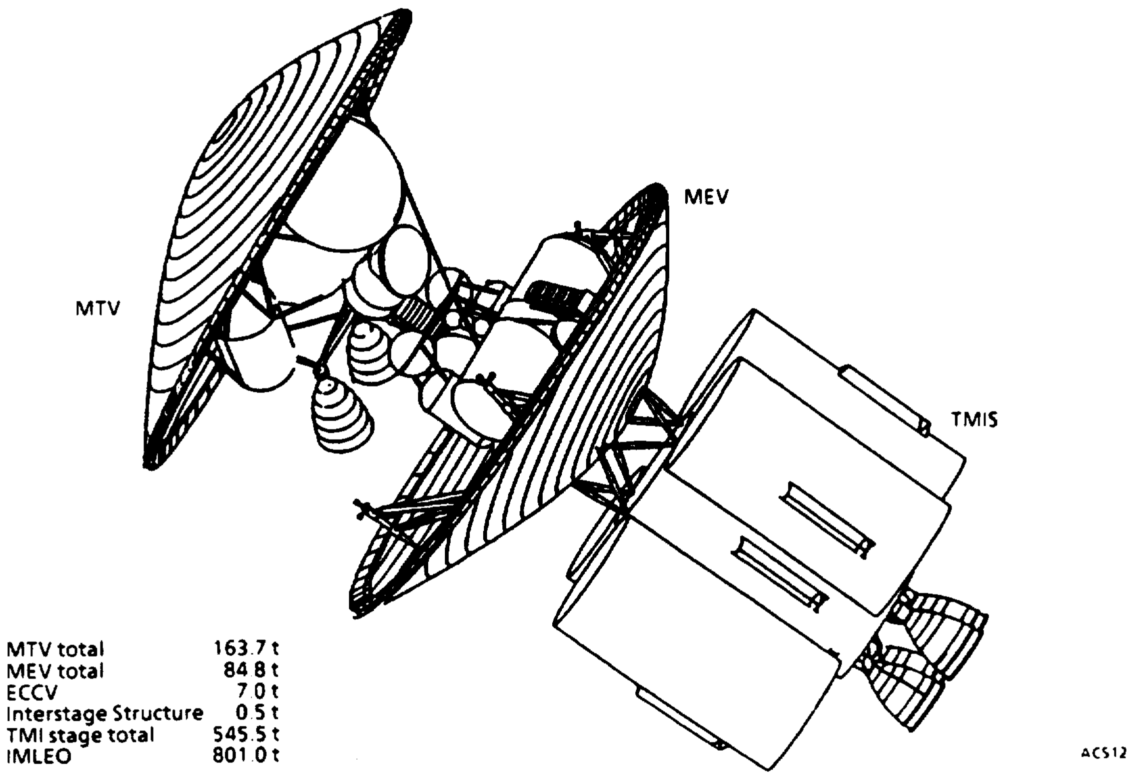

Figure 2-1. Cryogenic/Aerobraking Mars Mission Vehicle Ready for Launch from Earth Orbit

Only briefly mentioned in the "90-Day Study" final report, significant effort was also invested in conceptual definition of nuclear thermal and nuclear electric Mars transfer propulsion options. Somewhat lesser effort was spent on nuclear gas-core and solar electric concepts. Significant new technical work was accomplished on high-thrust and low-thrust Mars round trip trajectories and mission profile designs. Much of this work was performed by Boeing on the STCAEM contract supporting MSFC.

The "90-Day Study" was intended to collect the extant knowledge of how to conduct lunar and Mars exploration missions and to organize the information into a plausible and technically feasible overall program. While the "90-Day Study" has received considerable criticism, it accomplished its intended purpose quite successfully.

2.1.3 STCAEM Trades

At the completion of the "90-Day Study" technical effort, the Boeing STCAEM study returned to its primary objective of comprehensive tradeoff of Mars mission transportation analyses and trades. This activity had three main differences from the "90-Day Study" technical effort:

a. Three greatly different (range 10:1 people and cargo) levels of exploration activity were represented in mission scenarios in order to ascertain the sensitivity of results to overall scale of activity. A theme was established for each level:

1. Minimum satisfaction of SEI objectives as stated by President Bush,
2. Full menu of science, and
3. lunar industrialization and Mars settlement;

b. Technology advancement and evolution of mission activities were represented in each scenario; and

c. Tradeoffs emphasized life cycle cost and return on investment analyses rather than seeking minimum-mass solutions.

Seven Mars transportation architectures, summarized in figure 2-2, were played against the three scenarios, including development of life cycle transportation manifests. These architectures included the four main propulsion technologies, representation of in-situ propellants in the L2-based and direct modes, and representation of cycler orbits. (As described below, the STCAEM concept of "Mars direct" differed from the one later introduced by Zubrin and Baker.) Detailed performance and system mass calculations were made and configurations defined using computer-aided design tools.

Architecture	Features	Rationale
Cryogenic/aerobraking; all-propulsive option	Cryogenic chemical propulsion and aerobraking at Mars and Earth. LEO-based operations.	NASA 90-day study baseline; lower development cost
NEP	Nuclear-electric propulsion for Mars transfer; optionally for lunar cargo.	High performance of nuclear electric propulsion
SEP	Solar electric propulsion for Mars transfer; optionally for lunar cargo.	High efficiency of solar electric propulsion; find cost crossover for array costs.
NTR (nuclear rocket)	Nuclear rocket propulsion for Lunar and Mars transfer.	High Isp of nuclear rocket enables avoidance of high-energy aerocapture at Mars.
L2 Based cryogenic/ aerobraking	L2-based operations; use of lunar oxygen.	L2 base gets out of LEO debris environment. Lunar oxygen reduces resupply by ~factor 2.
Direct cryogenic/ aerobraking	Combined MTV/MEV refuels at Mars and LEO. "Fast" conjunction profiles.	Eliminates Mars orbit operations
Cycler orbits	Cycler orbit stations a la 1986 Space Commission report.	Eliminates boosting massive Mars transfer vehicle.

Figure 2-2. Seven Mars Transportation Architectures Investigated During STCAEM Trades

Summary of Results of STCAEM Trades - The trade studies compared transportation architectures on mass, trip time, reusability, and finally life cycle cost and internal rate of return, based on comparing life cycle costs streams for the alternatives. Figure 2-3 shows a representative mass versus trip time comparison of the transportation options for opposition mission profiles. The uncertainty band represents the range of performance requirements from "easy" to "hard" Mars opportunities; the comparison shown does not include Venus swingbys (see Section 2.3.3 below for discussion of mission profiles and their characteristics). Figure 2-4 presents a comparison of transportation

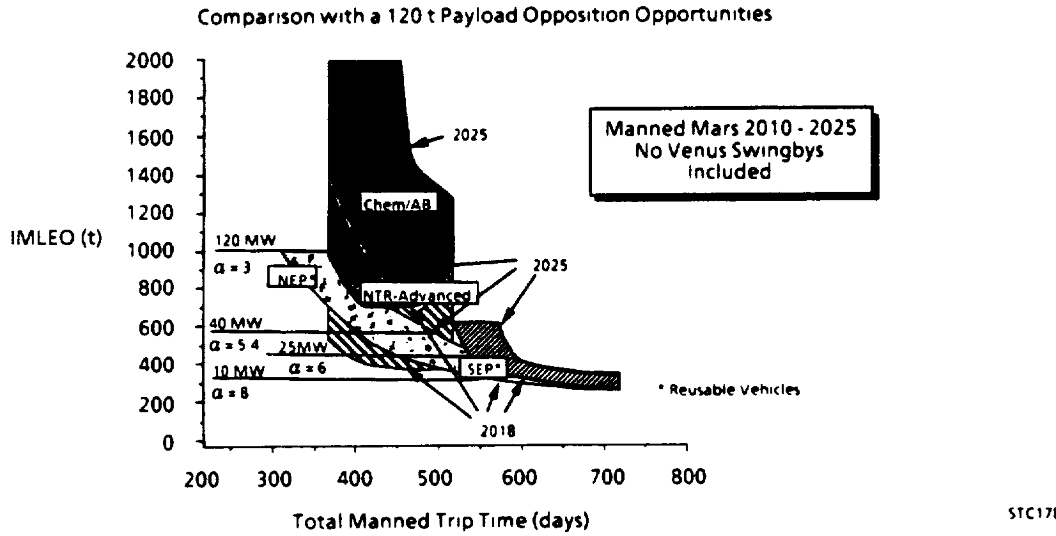

Figure 2-3. Propulsion Option Comparison for Opposition Missions

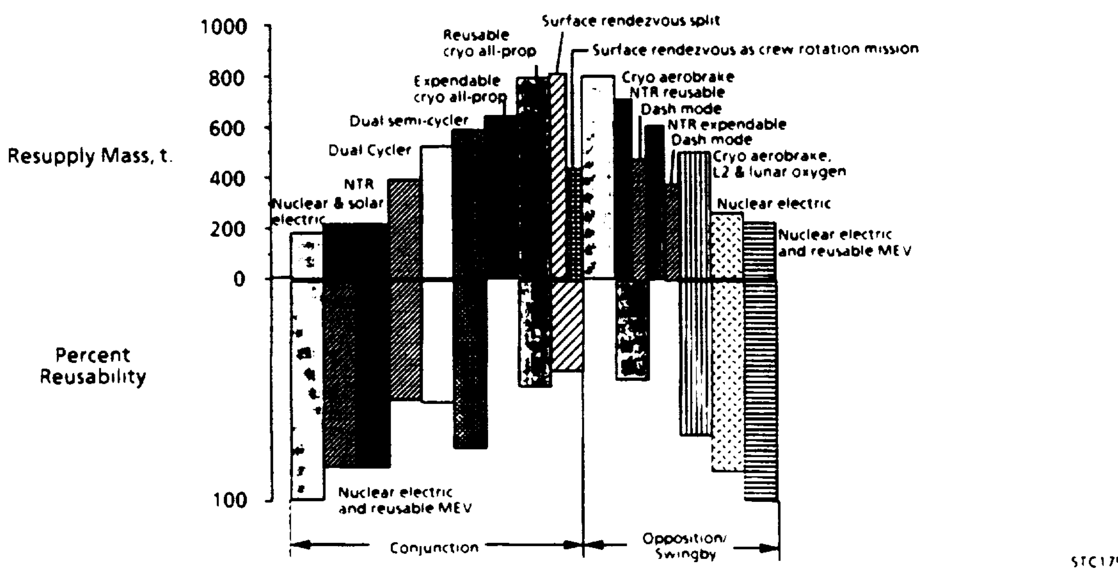

Figure 2-4. Architectures Comparison Summary

options in terms of "resupply" mass and reusability for opposition and conjunction profiles. In this case, Venus swingbys are used for the opposition profiles except for "easy" years where they offer little advantage. "Resupply" mass is the launch requirement (to low Earth orbit) needed to accomplish a typical mission; it takes credit for reuse of hardware on missions subsequent to the first one, which the usual "IMLEO" (initial mass in Earth orbit) comparison does not.

Trade Conclusions - The STCAEM trades progressed from these assessments to life cycle cost analyses and comparisons, including the internal rate of return (where applicable) for investment in a more advanced technology or system leading to lower cost later on. As mentioned above, transportation architectures were evaluated in total program scenarios over a wide range of activity levels. The conclusions presented below were primarily derived from the results of these cost and investment/return analyses.

Before discussing conclusions (most of which are still valid), one very important viewpoint shift between then and now needs to be pointed out. During the case studies, the "90-Day Study", and the Synthesis activities, widespread concern existed regarding the long duration of Mars missions and the risk inherent therein. This was evident in the one-year split sprint mission mentioned above and in the emphasis on fast opposition profiles in general. Since about the middle of 1992, the idea of using the Mars base itself as a safe haven has altered this view to the extent that the very first human mission to Mars is now posited as a conjunction type with surface stay of over 500 days. The significance of this is that electric propulsion systems are at a severe disadvantage under a requirement for fast trips. The conclusions reached by STCAEM in early 1991 change if fast opposition trips are not a requirement.

The STCAEM trade studies of 1990-91 reached many conclusions. Only those particularly relevant to the current Mars architecture assessment are summarized here; quotes are from the March 1991 Executive Summary Report:

Propulsion - "For a minimum Mars program, consisting of perhaps a half-dozen landings of a few days' stay each ... cryogenic all-propulsive minimum-energy missions ... are very attractive." A Mars-orbit-based conjunction profile with short surface sorties was presumed. The conclusion may also apply to a minimum program using a surface-based conjunction profile.

"The performance potential of a nuclear thermal rocket leads to less initial mass than cryogenic/aerobraking for most mission profiles. ... Return on investment tradeoff ... at the median activity level favored the nuclear rocket. ... The nuclear thermal rocket is indicated as the most economic and flexible Mars transfer propulsion system over a wide range of program activity levels."

The nuclear rocket concept presented in the report is shown in figure 2-5.

"While fast trips [referring to fast opposition trips a year or less duration] are technically interesting, they are probably not affordable in a space program with constrained funding."

"The inherently high reusability and low resupply mass of electric [propulsion] systems offers life-cycle cost advantages at high activity levels. Development cost for NEP and array production cost for SEP are major issues. ... SEP becomes very attractive at \$100/watt, showing about 10% return on investment versus NTR at the median activity level."

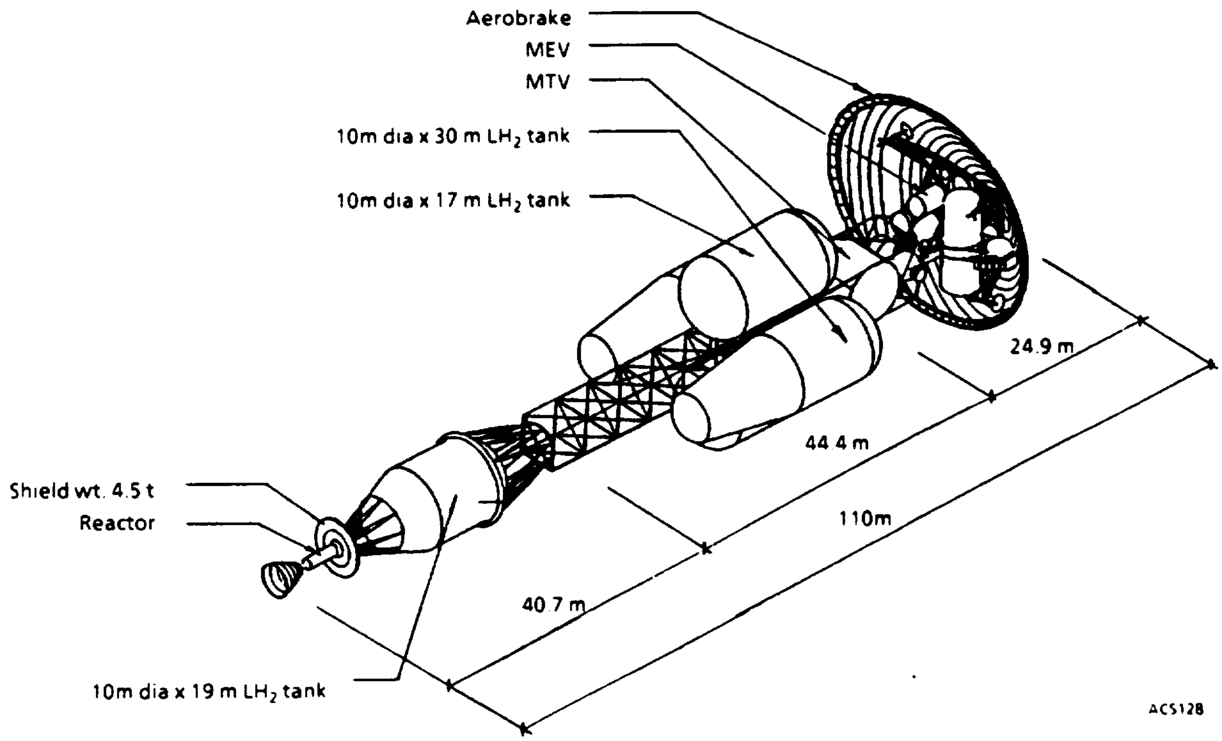

Figure 2-5. Nuclear Thermal Rocket Concept from STCAEM Phase 1

The L2-based, lunar oxygen architecture was found to have a poor return on investment in the oxygen production facilities. Subsequent study has indicated this conclusion to be sensitive to assumptions and implementation details; lunar resources concepts merit further investigation.

An extensive habitat tradeoff selected 7.6 meters diameter as preferred over 4.4 meters (space station diameter) and 10 meters.

Orbital Operations and Assembly - "Reviews and analyses of orbital assembly operations did not identify major problems."

"Benefits of very large launch vehicles appear not worth the cost."

It is important to note that when the STCAEM study began, the prevailing image of on-orbit assembly involved a very large and complex assembly facility that surrounded the entire Mars vehicle, and appeared to pose more of an assembly problem than the Mars vehicle itself. Concepts developed by STCAEM steadily evolved in the direction of lesser assembly facilities and reached a point by 1992 wherein the assembly "facility" consisted of a robotic device launched with the first section of the vehicle, as shown below in figure 2-6.

The issue of launch vehicle for SEI-type programs continues to evolve. Section 1.3 of Volume 2 of the current Phase 4 report presents a discussion of this.

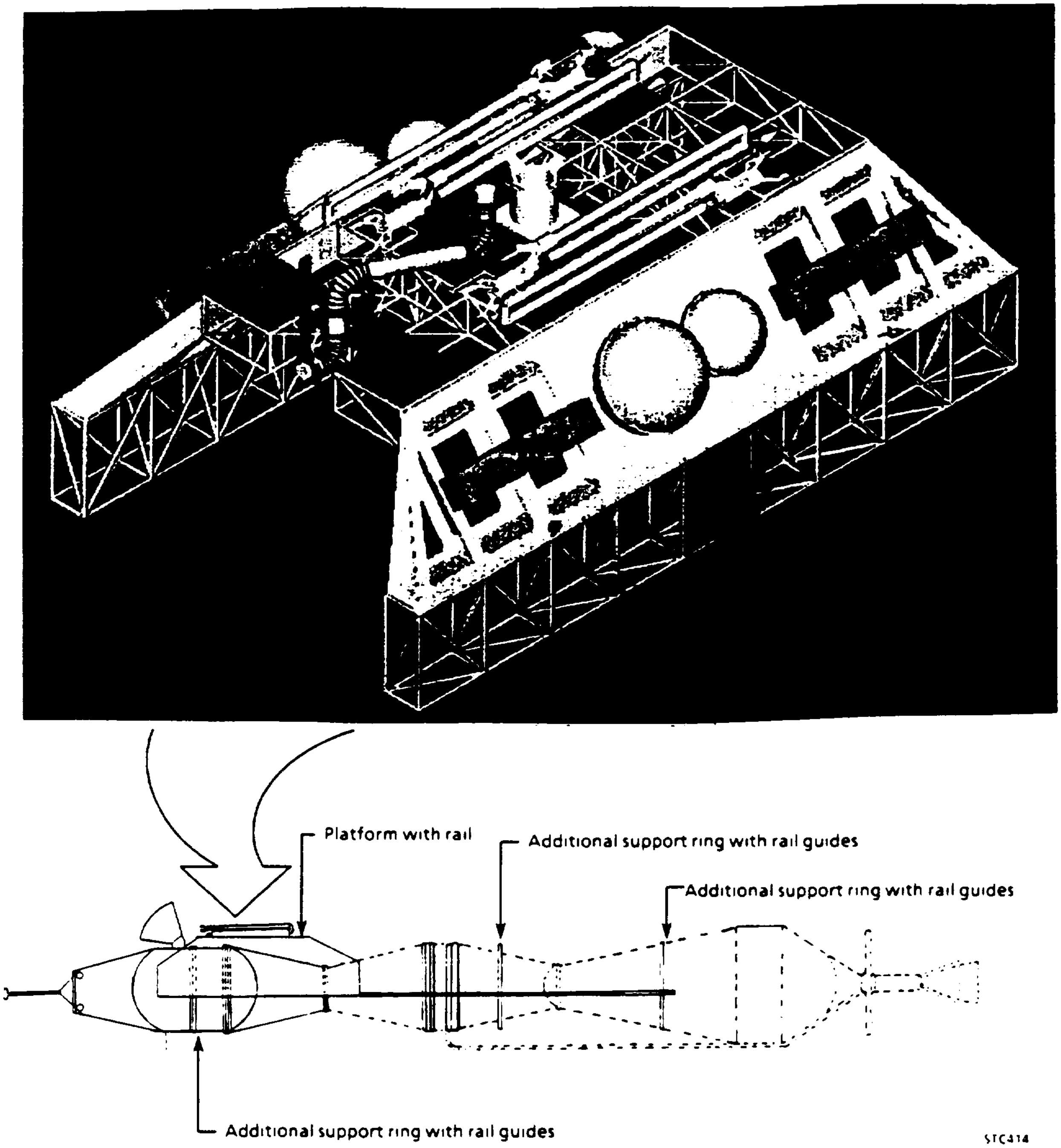

Figure 2-6. Assembly Device for Assembly of Nuclear Rocket Mars Vehicle (STCAEM Phase 2)

General Cost - The STCAEM trades posited a minimum program with much less life cycle cost than (roughly 1/3 of) the "90-Day Study". It of course included fewer and less ambitious missions. The STCAEM scenarios were inherently evolutionary in use of technology and mission characteristics. Lunar industrialization and Mars settlement were recognized as at least possible scenarios. Better definition of program purposes and functions were recognized as urgently needed.

"Before ultimate selection among these architectures can be made, better definition is needed as to the nature and activity level of the lunar and Mars exploration and development programs ..."

2.1.4 Outreach and Synthesis

"Sticker shock" reaction to the high estimated cost of the "90-Day Study" scenarios led to a call for innovation. We would have a national outreach for new ideas (we actually had two), and find "faster, better, cheaper" ways to send humans to the Moon and Mars. The outreaches did not discover very much that was genuinely new. This isn't surprising since space visionaries and planners have been thinking about Mars for over a generation, and communication in the community is quite open. The "Synthesis" report, *America at the Threshold,* gave little recognizance to the few innovations that were found. The Synthesis nuclear thermal propulsion baseline for Mars was very similar to the Mars missions recommended by the Agnew Commission of 1968, except for certain new technology items somewhat incidental to the main issues of transportation and habitation. Even the launch concept was a throwback to Saturn V technology. Life cycle costs were reduced somewhat relative to the "90-Day Study" by reducing the scope of the lunar base and lunar operations.

The Synthesis mission scenarios were a step forward from the "90-Day Study": more cost-efficient, better integrated, and directly derived from themes based on different rationale emphasis. The Mars scenarios progressed quickly from an initial opposition mission to the more productive long-stay conjunction profile.

The Synthesis report included a scathing, too-severe indictment of aerobraking as having excessive risk, and came down squarely in favor of nuclear thermal propulsion. Nuclear electric propulsion was mentioned as having potential.

Since the Synthesis report, a reduced-cost initial lunar program, the First Lunar Outpost (FLO), has been defined by NASA. Other proposals, generally not very credible, have presented even lower cost projections. Contemporary Mars studies, initiated by a workshop in Houston in August of 1992, are examining ways of reducing cost, mainly by minimizing Earth orbit operations in favor of Mars operations and by increasing commonality between lunar and Mars systems, especially habitats and planet ascent propulsion. One very high cost item, a 200+ tonne launch vehicle, was retained by the NASA studies, even though it creates a severe early program cost problem.

During the Synthesis period, the STCAEM study concentrated on refining the NTP concept by taking its definition one layer deeper. This included definition of subsystems and attention to the details of the on-orbit assembly problem. The resulting configuration is shown in figure 2-7; it could be assembled by a simple robotic fixture, as described earlier. An alternate concept requiring only berthing for assembly was also created; it is shown in figure 2-8.

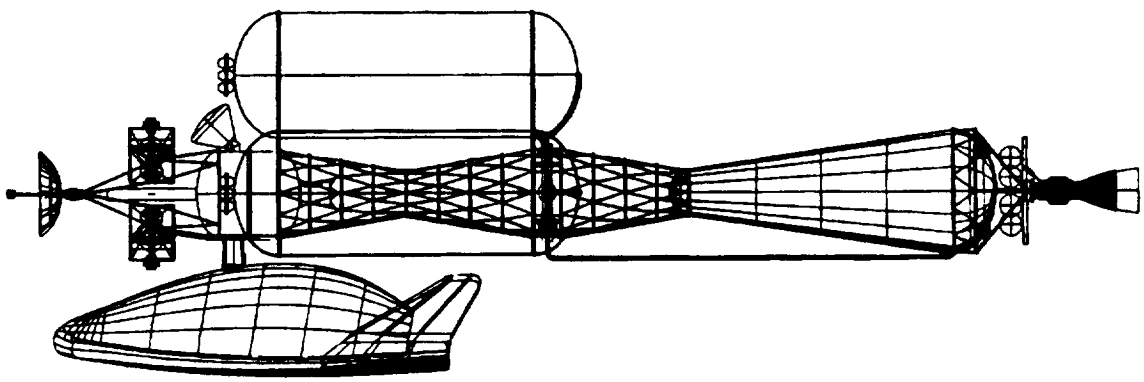

Figure 2-7. STCAEM Phase 2 Nuclear Thermal Rocket Vehicle Concept

The STCAEM analysis of Synthesis architectures examined abort profiles and philosophy in some depth. In particular, the Synthesis conjunction profile design approach was examined. This approach makes the Earth-to-Mars segment of a conjunction profile (after the first mission which is opposition) a segment compatible with an opposition-like Mars flyby and return to Earth, in order to provide an abort opportunity. The STCAEM conclusion was that this approach is good but doesn't go far enough. The likelihood of an abort was quantitatively estimated with the result that an abort after Mars orbit capture is more likely than during the Earth-Mars transfer. This means that the conjunction mission needed to be designed to fly an opposition profile abort, which in some opportunity years is a significant penalty.

The main activity of the STCAEM study during 1992 supported MSFC in development of the First Lunar Outpost (FLO) habitat system.

2.1.5 Mars Direct

The Mars direct profile as visualized by the STCAEM study was an evolutionary step beyond early missions; it consisted of an integrated reusable Mars transfer and landing system refueled on Mars with hydrogen and oxygen derived from in-situ propellants. The vehicle was to fly from Earth orbit to Mars landing on Earth-supplied propellants and from Mars surface to Earth orbit on Mars-supplied propellants. It was presumed that a Mars base complete with propellant production facilities had already been emplaced.

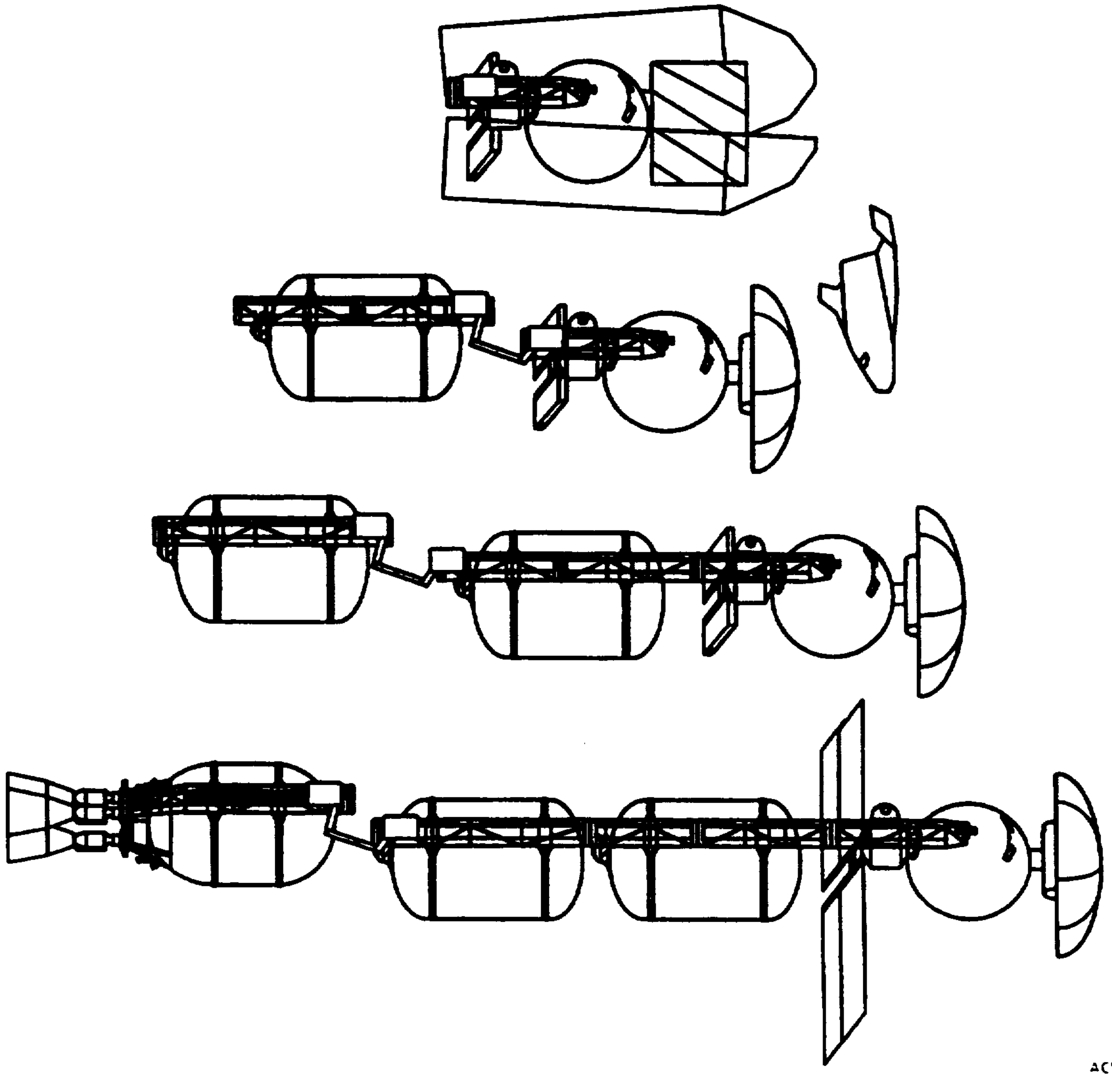

Figure 2-8. STCAEM In-Line Modular Nuclear Thermal Rocket Vehicle Concept

The Mars Direct profile popularized by Zubrin and Baker was the same fundamental profile but implemented very differently. Zubrin and Baker visualized an expendable-mode Mars direct profile as the most promising scheme for an early Mars mission and proposed one as early as 1999. In their scheme, an Earth return transfer propulsion and habitat system is prepositioned on Mars and refueled by methane and oxygen produced from Mars' atmosphere (with the aid of a modest amount of hydrogen brought from Earth) The refueling process uses automated propellant production and relatively simple robotics. The mission crew transfers to Mars, bringing their transfer habitat to Mars' surface via aerobraking for use on the surface during the 500-day stay. With a large enough launch vehicle, no Earth orbit assembly is required. Also, no Mars orbit operations are required.

The Zubrin/Baker scheme has been criticized as risky; also in order to reduce the size of the return system to be compatible with a single Earth launch, the return habitat as defined by Zubrin and Baker is extremely austere. Whether or not Mars Direct is adopted, it brings out important ideas:

a. Production of propellants from Mars' atmosphere is a greatly simpler proposition than production from lunar regolith.
b. An adequately robust habitation facility on the surface of Mars is more reachable as a safe haven than return to Earth for some portion of almost any Mars mission profile.
c. Restricting the first human mission to Mars to a short, opposition-profile stay is probably neither necessary nor desirable. (Zubrin's argument)
d. Architectures can be devised which make possible a first Mars mission much earlier than indicated by "conventional wisdom".

2.1.6 JSC Mars Working Group, 1992 - 93

In 1992, as SEI activities were winding down, a Mars Working Group was formed at JSC. The initial purpose of the group was to develop an architecture for beginning Mars exploration that would provide continuity and evolutionary context for the FLO mission definition activities then underway. After the 1992 election, it was clear that the SEI program as advocated by President Bush would not occur. The purpose of the group then shifted towards creating a basis for future evolution of space exploration. One of the ongoing activities of the group, before and after the election, was definition of a new Mars architecture approach.

In August of 1992, a meeting of the Working Group, including outside reviewers, focused on rationales for Mars missions. In addition to the usual science objectives, this meeting recognized an important function of Mars missions as assessing the future habitability of Mars for human settlement.

Architecture definition by the working group was strongly influenced by two factors: (a) The penalty of requiring the outgoing leg and orbit capture of a Mars conjunction profile to be compatible with opposition-type return to Earth, and (b) some of the concepts and ideas of the Mars Direct profile.

The Working Group undertook to define a Mars surface architecture that is sufficiently redundant and robust that it can serve as an abort safe haven, eliminating the need to design missions to always return to Earth to effect abort. It was not within scope of the STCAEM study to deal with surface architectures; it was simply taken as given that the surface architecture is adequate. STCAEM did perform analyses and review of the Working Group transportation architecture, as described in the following section.

2.2 STCAEM REVIEW OF JSC WORKING GROUP ARCHITECTURE

2.2.1 Architecture Summary

The Working Group's reference architecture employs three launches from Earth direct to Mars for cargo delivery and one additional launch for the crew mission the following opportunity. During the crew mission launch period, two cargo launches to support the following crew mission are also launched. The composite mission profile is diagramed in figure 2-9, taken from the JSC reference mission description. In addition to surface cargo, the cargo missions deliver the crew ascent vehicle and the Earth return vehicle. The former is fueled with in-situ propellants after landing (but prior to the crew departing Earth) and the latter is parked in Mars orbit, fully fueled, awaiting use at the completion of the crew mission. Both the crew ascent vehicle and the Earth return vehicle use methane and oxygen propellants. Methane was selected for the ascent vehicle because this provides an 18:1 gain for the hydrogen delivered whereas simply making oxygen and using hydrogen from Earth provides 7:1 gain. Methane was selected for the Earth return for engine/propulsion commonality with the ascent vehicle and because the storage temperature for methane is about 70 K warmer than for hydrogen.

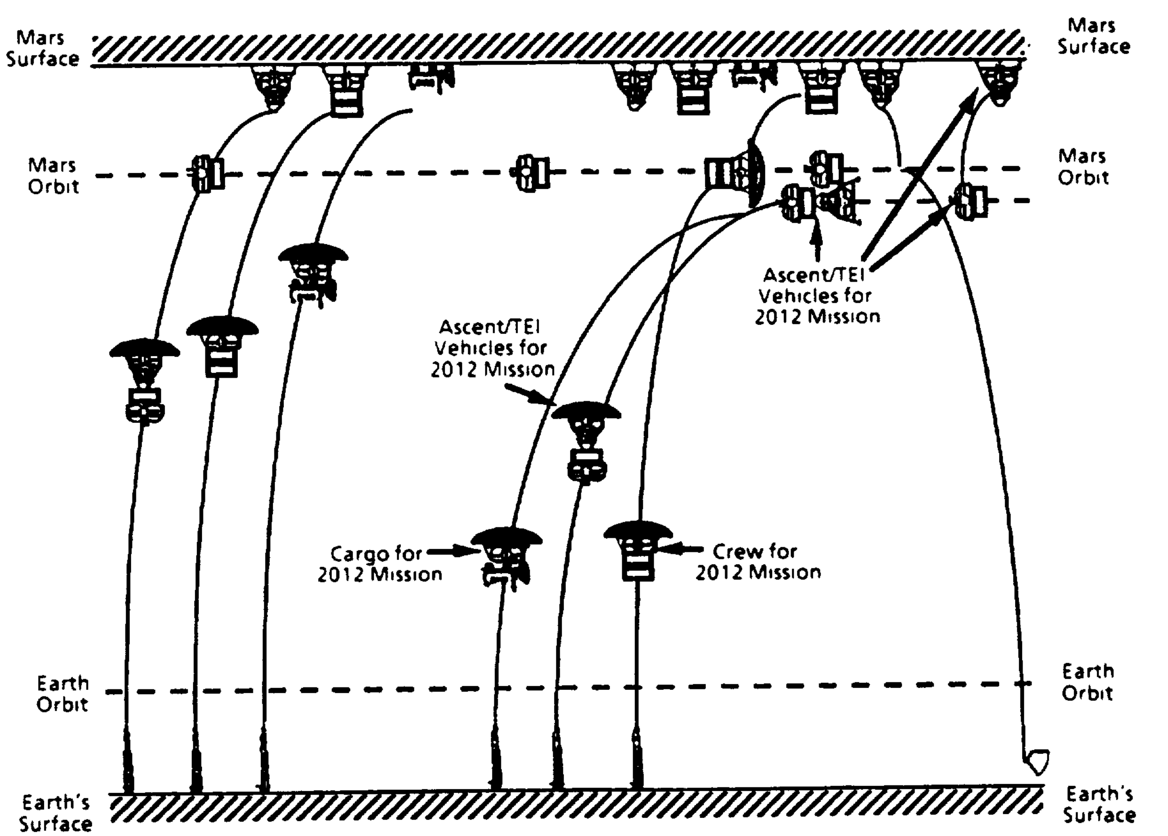

Figure 2-9. Composite Mission Profile for the JSC Working Group Mars Mission

The crew mission also goes on a direct launch from Earth. The mission vehicle includes an aerobrake for Mars arrival, a transfer/surface habitat, other small surface cargo such as a rover, and a landing stage. The transfer habitat goes to the surface to become part of the surface infrastructure. The habitat for the return transfer is already parked in Mars orbit before the crew leaves Earth because it is part of the Earth return vehicle. The Earth return vehicle and the ascent vehicle for the following mission opportunity are delivered to Mars as cargo, arriving there about the same time as the current crew. These are available as backup for the return trip.

Several variations on the reference architecture were described, but all operated on the same basic concept. The review provided here is generally applicable to these variations.

2.2.2 Review Viewpoint

What one looks for in any new architecture is advances in understanding as well as improved features and characteristics. Weaknesses and drawbacks are somewhat less important provided they are identified and flagged for future work. This review identifies both with the idea that the current reference architecture is the latest word, but not the last word, in Mars architecture development. Forthcoming studies, it is presumed, will continue to make improvements.

2.2.3 Plaudits for the Reference Architecture

Abort - The reference architecture successfully challenges the unstated assumption that "abort" always means return to Earth. Adopting the notion of an adequate safe haven on Mars eliminates severe constraints on certain portions of the mission profile design. However, this architecture relies almost entirely on abort to Mars surface.

Fast Opposition - Earlier architecture studies, including the STCAEM trades described above, were distorted by considering propulsion system performance potential for "fast opposition" profiles as a primary evaluation criterion to the point that systems such as electric propulsion were not optimized to show their strong points to good advantage. The reference architecture removes that emphasis by pointing out the inherent advantage of the conjunction profile: an order of magnitude more stay time on Mars for the investment in each human mission. Removing this emphasis also eases electric propulsion performance requirements.

Surface Systems - Inherent in the conceptual advances above is the idea of a robust and redundant surface architecture that can be depended on as a safe haven. While the STCAEM study was not involved in Mars surface systems, obtaining a realistic definition of surface activities and surface systems is essential to the evolution of transportation architectures because of abort significance as well as understanding cargo delivery requirements. The reference architecture made important contributions in this area.

In-Situ **Materials** - It almost goes without saying that use of in-situ materials is important to any efficient architecture with long-range potential. The reference architecture places emphasis on early use of in-situ materials; this is especially important since there has been a tendency in the technology community to think of in-situ materials technology as something that can comfortably be deferred. The long-range evolution of Mars architectures may well be paced by the development of in-situ materials technology.

2.2.4 Critiques of the Reference Architecture

Abort - While the reference architecture made a significant step forward in understanding Mars mission aborts, the arrangement of the architecture eliminates several abort modes that should not be eliminated. Two examples are: (a) Abort from a failure during trans-Mars injection is eliminated because the crew mission vehicle does not include an Earth entry module; (b) Abort from the descent to Mars, or after a landing in the wrong location, is eliminated because the descent system does not include an ascent vehicle. Also, if an ascent were made to the Earth return vehicle, it does not have sufficient consumables to sustain the crew until an Earth return opportunity.

The abort analysis is described in detail in Section 3 of this volume.

Growth and Evolution - In some ways the reference architecture is commendable in this area; for example, it inherently adds to the surface infrastructure on every mission. However, it gives very little consideration to the long-range evolution of Mars surface operations and transportation technology. By not considering these evolutions, it seems to presume that they will not occur, or at least makes no provisions for them.

Problems and Penalties - Certain aspects of the architecture appear motivated primarily to eliminate problems for which the "cure" seems worse than the disease. Eliminating Earth orbit operations is a prime example. Earth orbit assembly does not seem to be much of a problem unless it involves extensive EVA. The division of the crew mission systems which causes the abort problems noted above is driven by the need to divide the mission into equal trans-Mars masses to suit the direct launch constraint.

A second example is the use of methane in-situ propellant. In-situ propellant is essential to the workability of the Mars Direct architecture; without it, launching the Earth return habitat from the surface of Mars to its return trajectory would require a completely impractical Mars landing mass. Both methane and hydrogen have advantages and disadvantages in this application and which is the best propellant has not been conclusively demonstrated. In the present reference architecture, in-situ methane serves only to fuel the ascent vehicle. The increase in landing mass without in-situ propellant is about 20%. Use of hydrogen, with its higher Isp, in the Earth return stage parked in Mars orbit would *decrease* the delivery mass.

Technology Menu - The architecture introduces five new technologies which must be regarded as risky from the program management point of view: (a) nuclear thermal propulsion; (b) aerocapture at Mars; (c) nuclear electric power for Mars surface application; (d) in-situ propellant production; and (e) robotic assembly of the initial base under conditions of severe communications time delay. (Shuttle had three - its engine, its TPS, and its data management system.)

It is recommended that consideration be given to (a) eliminating either nuclear propulsion or aerocapture; (b) evaluating whether solar/regenerable fuel cell energy can be used on the first mission, i.e., with natural sunlight for the bio-chamber and day-only operation for in-situ propellant production; (c) restricting use of in-situ propellant production to fueling surface rovers, with growth to ascent vehicle use; and (d) evaluating whether a surface base can be devised which is usable in as-landed condition if necessary. Required robotic assembly is no more than the surface mobility needed to move base elements to within reasonable proximity to one another. Interconnection would be attempted as an enhancing feature; (e) Devising a surface base and mission architecture that can sustain failure of any one Earth launch.

2.3 CURRENT ISSUES

The issues evaluated in this section respond to the STCAEM statements of work for Technical Directives 16 and 17. These issues have, for the most part, not been in the mainstream of analyses and trades from prior architecture studies. Taken together with the review of the JSC Working Group architecture in the previous section, these provide the basis for the general review of Mars architectures in the following section.

2.3.1 Evolution to Crew Rotation and Resupply Operations

Conjunction-class visits to Mars are logical for the first few human missions whether the reference architecture or another is used. If one presumes that the purposes of Mars exploration lead to permanent surface operations, a change in profile may be needed. Sequences of conjunction profile missions leave gaps of several months in presence, unless crew members stay for more than one synodic period, as illustrated in figure 2-10. As also illustrated in the figure, sequences of opposition profiles permit a regular crew rotation and resupply operation with continuous occupancy of a surface base. This offers a logical step between a base phase and a settlement in which people may stay indefinitely.

Valid reasons exist for permanent human presence on Mars whether or not a settlement phase is initiated. Certain scientific research benefits from long-term continuous operations. Gaps in base occupancy with a sequence of conjunction missions may occur at scientifically inopportune times. In-situ food growth is expected to reduce

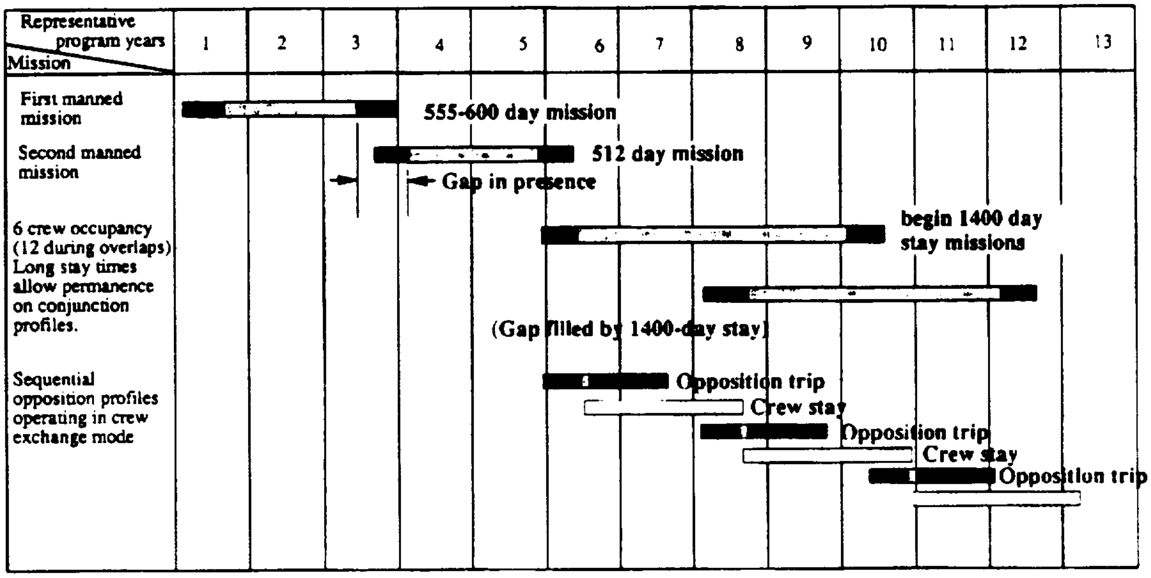

Figure 2-10. Achieving Permanent Presence on Mars

subsistence resupply cargo requirements by half or more; operation of a bioregenerative life support system will be seriously compromised by periodic shutdown and restart.

Direct subsistence requirements for a crew, assuming a life support system that regenerates all water and breathing atmosphere, include food and life support consumables such as filters, hygiene items and medical supplies. The mass requirement is about 4 kg per person per day, of which more than half is food. A bioregenerative system is expected to evolve to essentially complete food supply capability, but should be backed up by an adequate emergency reserve food supply (3 years recommended) before it is permitted to be critical to survival. The recommended approach to this is to overproduce long-term storable foodstuffs with the bioregenerative system until the reserve is built up, and to consume and replenish the reserve regularly so that food residence time in the reserve does not exceed the 3 years' emergency supply. For a six-person base, the direct subsistence resupply for one synodic period is slightly less than 20 t., dropping to less than 10 t. when bioregenerative food production reaches full capacity. Figures 2-11 and 2-12 compare resupply scenarios with and without a bioregenerative food supply (called bio-chamber in the Figures). These scenarios are for conjunction sequences with crew stay durations of almost 4 years. Figures 2-13 and 2-14 compare supply inventories for these scenarios. With the bio-chamber, one 50-t. cargo flight per opportunity delivers the needed subsistence supplies and enough capacity for current estimates of spares (see below). With out the bio-chamber, one cargo flight does not quite satisfy subsistence requirements. Section 5 of this volume presents additional information on these results.

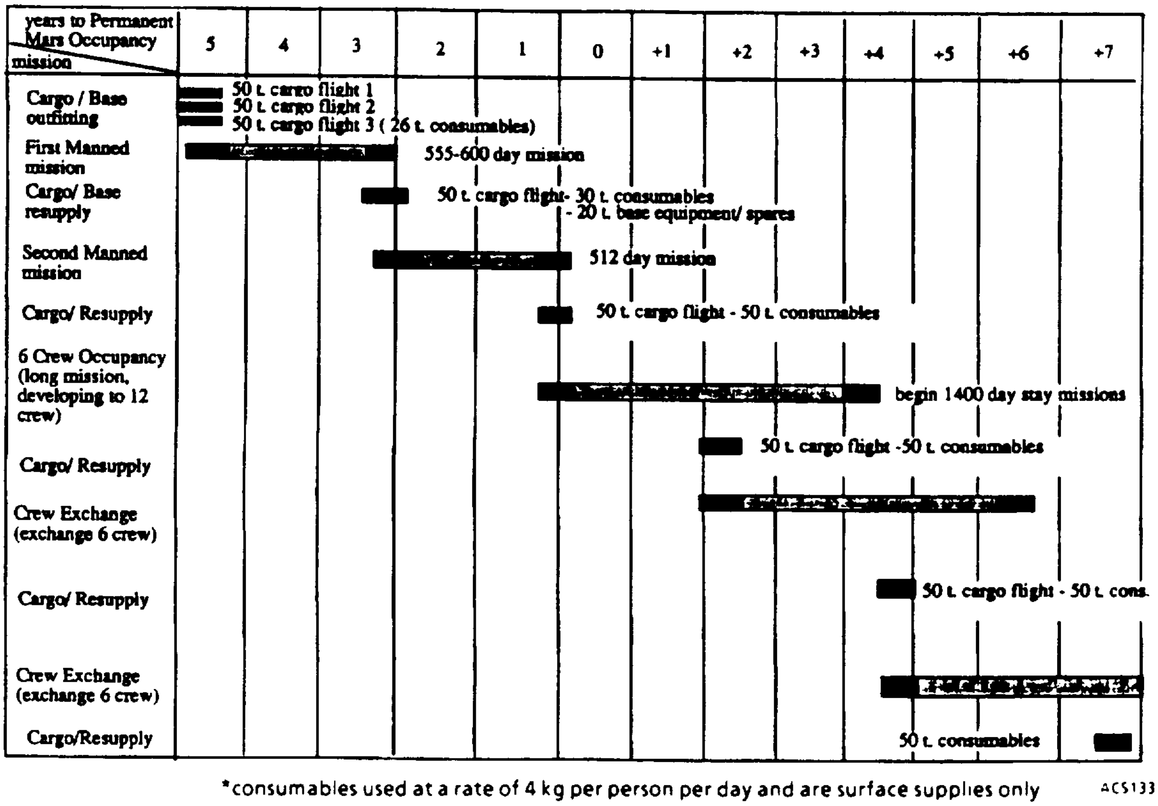

Figure 2-11. Mars Resupply and Evolution, Surface Consumables without Bio-Chamber Support

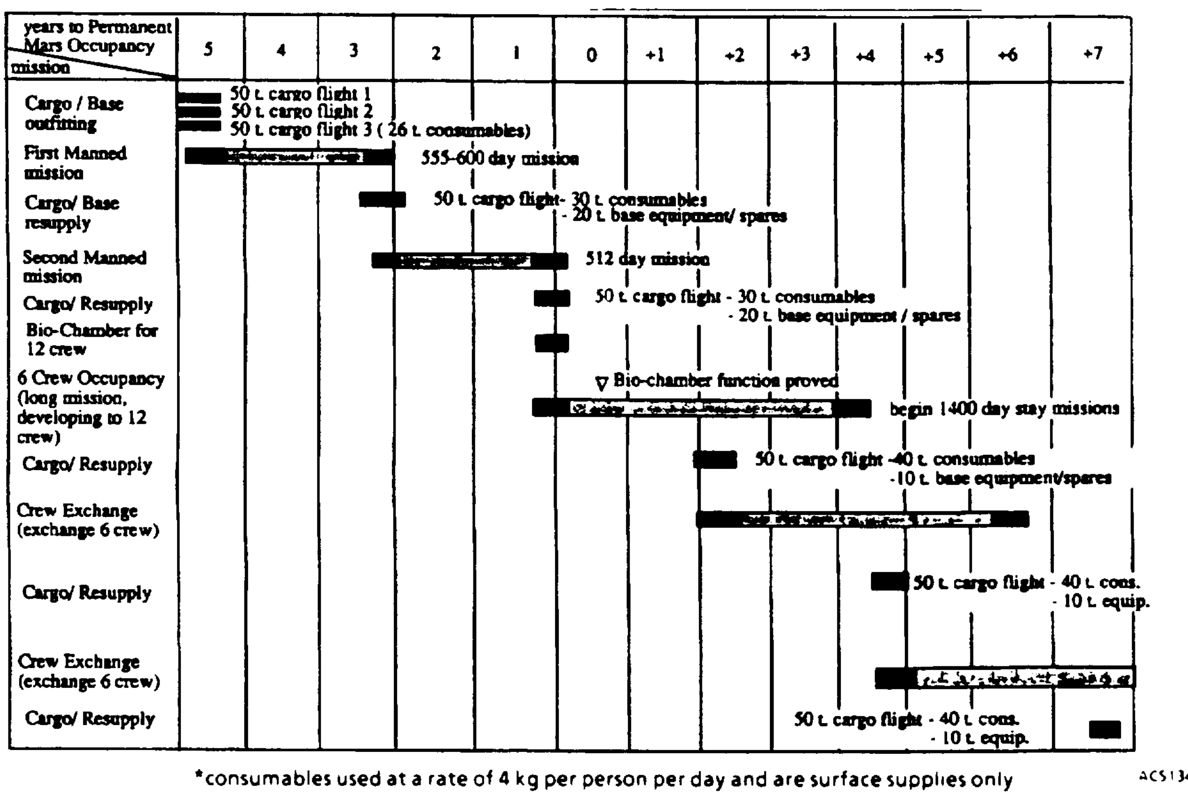

Figure 2-12. Mars Resupply and Evolution, Surface Consumables with Bio-Chamber Support

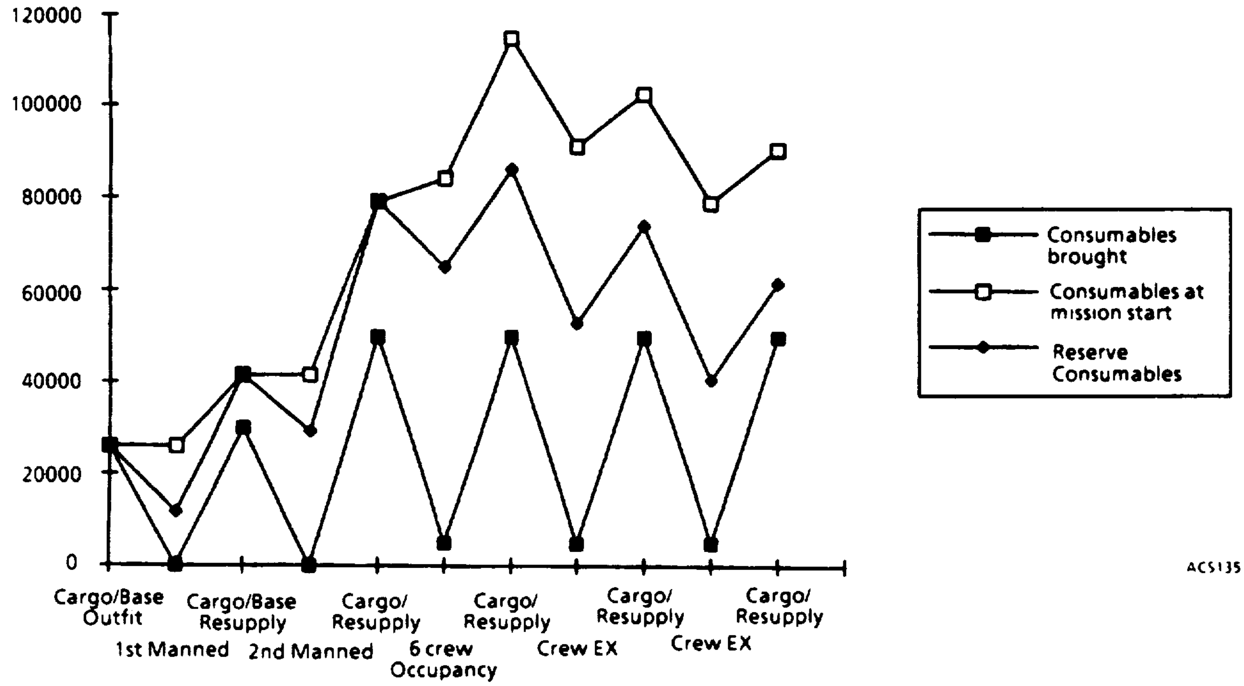

Figure 2-13. Consumables Resupply and Inventory History without Bio-Chamber

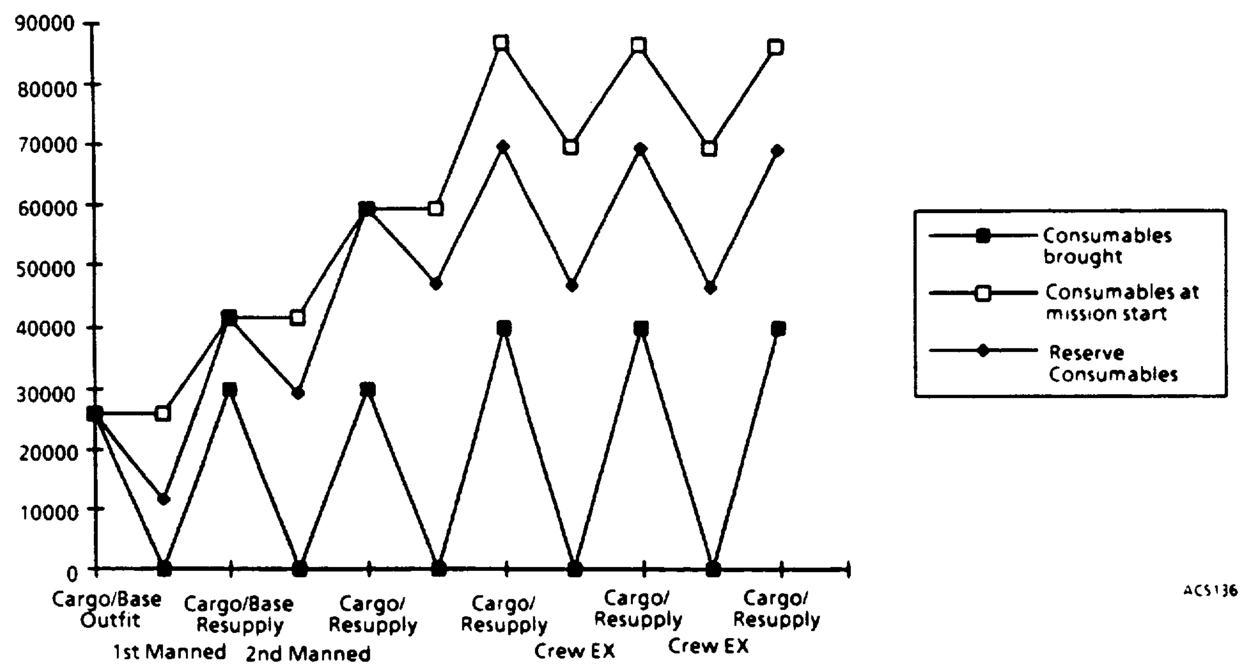

Figure 2-14. Consumables Resupply and Inventory History with Bio-Chamber

Additional resupply is required for base systems spares and for scientific and technological operations. Valid estimates do not exist for either of these. STCAEM has used 2% of gross mass per year for spares, which comes to 6 to 7 t. per synodic period for a six-person base. Masses for scientific and technological operations depend entirely on the nature of the operations, not yet defined.

Energy for base and surface transport operations must be regenerable or nuclear, or surface transport will be extremely limited. Nuclear plants will require replacement on intervals probably no greater than 20 years and possibly much less. A solar/RFC system might get by with depot-level (piece-part) repair of fuel cells and electrolyzers, in which case its resupply would be less of a burden than nuclear, but the initial installation will be about a football field of solar arrays (for 80 kW average)* and mass of RFC equipment more or less equivalent to the mass of a nuclear power system.

*(Solar flux = 525 Watts/m^2)(Atmosphere transmission 0.8)(day/night 0.5)(non-tracking $2/\pi$)(array efficiency 0.15)(storage net efficiency 0.8) = 16 W/m^2.

Low and high thrust systems fly trajectories which are similar from the overall architecture point of view; that is, either system can perform opposition-like and conjunction-like profiles. Therefore, the discussion of transition to crew rotation and resupply applies to either propulsion technology. As described below in section 2.3.3, low thrust systems are more sensitive to reductions in transfer time than high thrust systems; therefore, low thrust systems must be specifically assessed against desired transfer time constraints.

2.3.2 In-Situ Production of Propellants on the Moon and Mars

Lunar oxygen - The potential value of producing oxygen from the lunar regolith has been recognized since the 1960s. Lunar rocks are about half oxygen, but this is a new process field since oxygen is readily available on Earth from the atmosphere. The O'Neill space colonization proposals stirred up academic interest in specific production processes, and dozens of candidate processes have been identified. Many have been demonstrated on a laboratory scale, a few with actual lunar materials from the Apollo samples.

The STCAEM study as well as many others identified benefits of using lunar oxygen in an Earth-Moon transportation system. Relevant to this report is the use of lunar oxygen in Mars transportation, also considered by the study. The candidate lunar oxygen architecture presumed delivery of lunar oxygen to a transportation node at lunar libration point L2, where it is used to supply a cryogenic/aerobraking Mars transfer vehicle with oxygen. Hydrogen is supplied from Earth, and in this architecture, hydrogen from Earth is also used in the cryogenic lunar transportation system between the lunar surface and L2. This transportation system obtains its oxygen on the lunar surface and its hydrogen at L2, where it is delivered from Earth. Mars cargo payloads and crews are delivered to L2 from Earth by a lunar-type transportation system

Somewhat surprisingly, even if hydrogen must be supplied from Earth, use of lunar oxygen saves as much as 300 t. equivalent mass in low Earth orbit for each mission compared to basing in low Earth orbit. When the estimated mass delivery requirements and cost of emplacing the lunar oxygen production system are considered, the time required to recoup the investment in lunar oxygen production is several Mars opportunities. The return on the investment in lunar infrastructures is poor.

Lunar Hydrogen Production - Later in the study it was considered that production of propellant for a nuclear thermal propulsion system could become an attractive option, if feasible. The performance of a nuclear rocket is poor with any propellant other than hydrogen. Hydrogen is, of course, present in the lunar regolith as a result of solar wind implantation. Producibility is generally viewed as poor, since concentration is only 50 parts per million by mass. (This concentration estimate comes from analysis of Apollo lunar samples. The hydrogen is driven off by heating the bulk regolith.) Given the representative regolith density of 2000 kg/m^3, the density of hydrogen is about 0.1 kg/m^3, somewhat greater than the density of hydrogen gas at STP. This suggests that producibility might be acceptable.

Accordingly, a brief study was made to obtain a crude estimate of the mining and production system needed to extract hydrogen from the regolith. The analysis was mainly assumptional but served to generate very preliminary estimates of equipment size, mass and cost. These estimates in turn were used to estimate the economic feasibility of lunar hydrogen production as a source of nuclear rocket propellant for Mars missions.

Estimates of required propellant production on the Moon were based on the L2 basing concept described below in Section 2.3.3. It was determined that a hydrogen production rate of 100 t. per year is suitable for rough sizing of a production system. Results are summarized in figures 2-15 and 2-16. Assuming nuclear power, and that thermal heat for evolution of hydrogen from the regolith would be delivered directly from the reactors rather than by means of electrical generation, the mass of production equipment required on the Moon is about 600 t., not including a lunar habitat system which might be needed to support maintenance and operations personnel. IR&D studies of other applications of lunar hydrogen indicated that the mass and support requirements of a lunar habitat systems are much less important than the mass of the hydrogen production equipment itself. The payoff time for lunar hydrogen as a propellant supply for a Mars nuclear thermal propulsion system is about 3 Mars opportunities. While the estimates of lunar hydrogen production facilities are very crude, this approach is worthy of further study as an evolutionary goal for low-cost Mars transfer propulsion.

Lunar Electric Propulsion Propellant - Lunar hydrogen or water produced from lunar hydrogen and oxygen could be used in an electric propulsion system. Current ion engine technology uses heavier noble gases (argon to xenon), heavy alkali metals or mercury as propellants. All are very scarce on the Moon. Light alkali metals, especially sodium, are

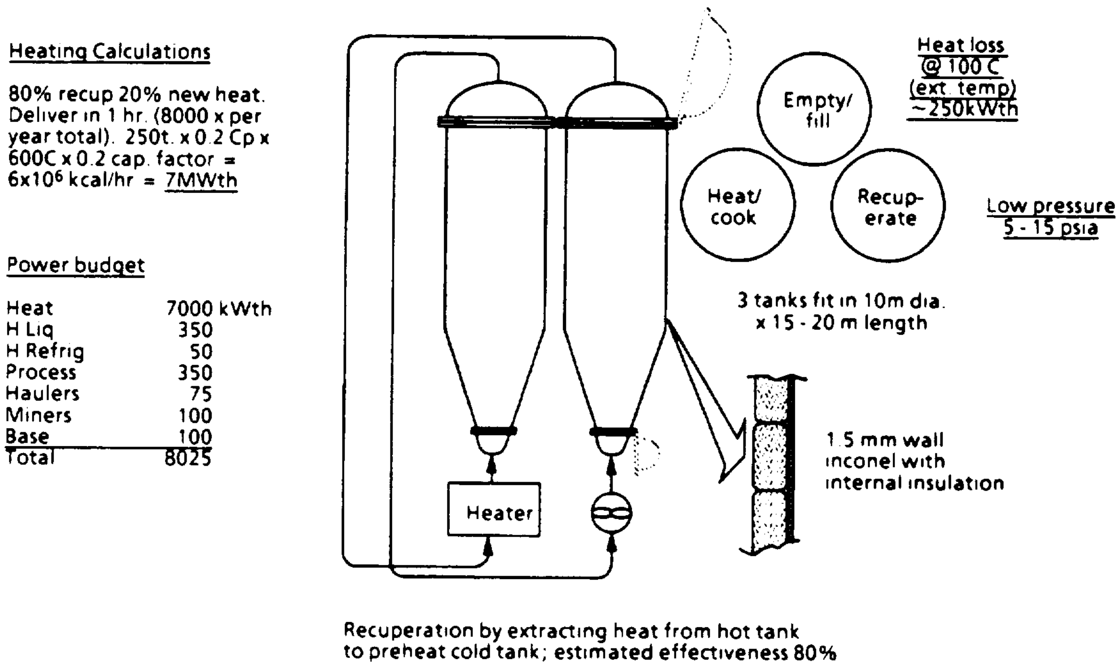

Figure 2-15. Hydrogen Cookers

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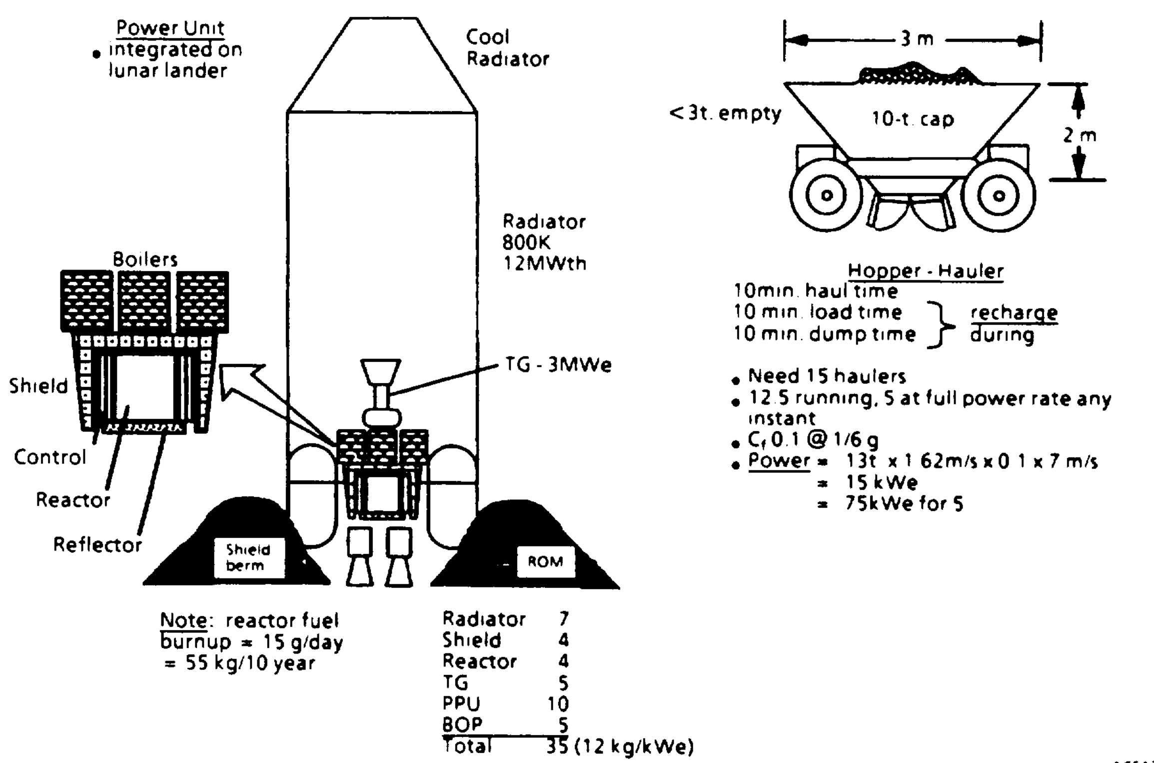

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Figure 2-16. Lunar Hydrogen Infrastructure Elements not as scarce. Plasma thruster technology can use hydrogen and might use water. Some pulsed plasma devices might eventually use almost anything. Presently, lunar hydrogen appears to be the best bet based on state of the art of plasma thrusters and availability of material.

Asteroidal Volatiles - Recently (August 1992), volatiles were discovered evolving from a near-Earth asteroid on photo plates taken sometime earlier. (The object was at that time identified as a small comet.) Long-term dynamical simulations of the motions of asteroidal and cometary bodies indicate that near-Earth asteroids come from the outer solar system "deep freeze" by way of planetary gravity assists; many may have been in the inner solar system for periods on the order of "only" 10^7 - 10^8 years. Objects with average distance from the Sun 2 - 4 au probably retain interior volatiles over such periods by formation of highly insulating surface layers of dust and slag. These objects could be economic sources of water and perhaps other materials for propellant and other uses.

This line of reasoning suggests that Mars' moons have been at Mars too long to be good sources of volatiles.

The flight mechanics of access to these objects is similar to that for Venus and Mars; windows of opportunity exist when an object has a near encounter with the Earth. For an object with aphelion great enough to retain water over long periods, a window of opportunity occurs every few years and the delta V to rendezvous with it is in the range 6 - 8 km/sec outbound and inbound; most of each delta V occurs departing and returning to Earth. Identification of several such objects as volatiles carriers (there are hundreds known to be in such orbits) would yield enough transfer opportunities to create a viable resource.

Considerations of trip time, risk and cost cause us to think of robotic mining vehicles as the way to acquire propellant from these asteroids. A typical mission profile departs Earth during a near approach, reaches the asteroid several months to a year or so later, spends several months to more than a year at the asteroid extracting volatiles, and a similar period returning to Earth. Two vehicle types have been proposed. The first, by Zuppero and others, is a nuclear thermal water rocket with moderate Isp (~400) and very low-mass, large water tanks. High delta Vs are attainable because of a presumed very high mass ratio capability due to the low-mass water tanks. The other, proposed by G. Woodcock, argues that the mass of mining and power equipment as well as practical tank design considerations would reduce the mass ratio readily attainable. It uses nuclear electric propulsion; the nuclear electric source is also useful to support mining operations. A top-level economic analysis of the latter concept indicates that *if* practical amounts of volatiles are available from several such asteroid sources, this in-space source of propellant is likely to be more economic than lunar surface sources. In either case, the best Earth-vicinity depot location is the lunar L2 libration point. Asteroidal sources would enable Mars orbit also to be used as a depot location.

Mars Atmosphere as a Source - Mars' atmosphere consists mainly of carbon dioxide. Three means of propellant production are possible. One, identified several years ago, is to dissociate carbon dioxide into carbon monoxide and oxygen. Both can be liquefied and burned in a rocket engine with estimated specific impulse about 250 seconds. The second is to simply use the oxygen from this dissociation process with hydrogen or other fuel brought from Earth. The third is to react hydrogen brought from Earth with carbon dioxide to produce methane. Oxygen is a byproduct, and additional oxygen can be produced to obtain the optimum mixture ratio for an oxygen-methane rocket engine. All of these processes have at least been demonstrated in laboratory equipment on a reasonable scale, and some are industrial processes on Earth.

While Mars' atmosphere is tenuous, typically 0.01 kg/m^3 at the surface, a simple calculation indicates that reasonable production rates should be achievable without difficulty. Imagine a 10-cm inlet pipe with flow velocity 10 m/sec. The flow rate is 7.85 x 10-4 kg/sec (of Mars atmosphere), which is 15 t. in a little more than 200 days. Production rates relative to atmosphere inflow depend on what is being produced but are roughly comparable to atmosphere inflow. Typical scenarios for MEV operations involve production rates from 10 to 50 t. per year. Power consumption also varies with specific product but is typically on the order of 1 kWe per ton per year. It is clear that production of propellant from Mars atmosphere is not nearly as technically challenging as production of oxygen or hydrogen from lunar regolith.

Some reusable MEV scenarios require all propellant to be obtained from Mars. Also, the propulsion performance requirement for a reusable MEV demands at least the 375 Isp of oxygen-methane and is better satisfied with oxygen-hydrogen performance. Mars' regolith presumably has no hydrogen. However, it is anticipated that substantial amounts of (frozen?) water will be found on Mars. Hydrogen and oxygen can readily be produced from water by electrolysis. Power needs are somewhat more than for the atmosphere production described above. The main issue is that we don't know where on Mars or in what form water will be discovered. There is almost certainly water in the polar caps but this is a very inconvenient location. Further assessment of water as a propellant source needs more information on water availability on Mars.

Potential applications of Mars propellant are: (a) Fueling of Mars ascent vehicles as in the JSC 1993 "reference architecture", (b) Fueling of reusable Mars excursion vehicles (MEVs) as in one of the STCAEM seven architectures, (c) fueling of complete Earth return vehicles as in Zubrin's Mars Direct architecture, and (d) fueling of surface mobility vehicles for Mars surface operations. The delta V for ascent from Mars to Mars orbit appears to make refueling of orbit-based Earth return vehicles impractical, but no thorough analysis has been done.

2.3.3 Review and Update of Alternate Mission Profiles and Modes

This review provides an overview of current knowledge and architecture design results for the alternative mission profiles applicable to human Mars missions. All presently known profiles and architectures can be collected into the five groups represented here.

Generic Conjunction Options. The synchronism of the conjunction profile employs long wait times at Mars (normally more than a year) to access low-energy transfers from Earth to Mars and Mars to Earth. The space vehicle makes one fewer revolutions around the Sun than Earth, i. e. about 1-1/2 vs. 2-1/2. A conjunction profile may use relatively high energies to obtain fast trips, but each transfer is near the minimum energy for the particular trip time.

Chemical (usually cryogenic), chemical/aerobraking, nuclear thermal, and nuclear and solar electric propulsion systems are all represented here. The generic conjunction profile departs Earth, arrives at Mars by capture into a Mars orbit, executes a landing using a landing craft (not the entire vehicle), uses a portion of the landing craft for an ascent to rendezvous with the orbiting craft, and the orbiting craft is used for return to Earth from Mars orbit. Many variations are possible, such as the JSC 1993 reference architecture. It falls into this category because it uses conjunction-type transfers and synchronism and because the craft for return to Earth is parked in Mars orbit.

The wait time in Mars orbit is enough that planetary oblateness perturbation of the orbit line of nodes and line of apsides can be used to advantage for orbital alignment. A high-thrust conjunction profile can obtain very nearly the full advantage of an elliptic parking orbit at Mars. By properly selecting the orbit inclination and period, the lines of nodes and apsides can be in near-ideal alignment for arrival and departure. Consequently, conjunction profiles are marked by low Mars capture and departure delta Vs, on the order of 1200 m/sec for moderate transfer times. Orbital alignment is not an issue for low-thrust profiles.

The lower delta Vs for conjunction profiles are somewhat offset by the greater consumables requirements for the longer total mission durations. Consumables, however, are outweighed by the propulsion requirements of higher delta V for opposition profiles. The extra consumables for the roughly 500 days' greater duration are about 20% of a mission habitat system mass (even if the mission habitat must carry consumables for the entire duration, as may be the case when abort requirements are included), whereas in the case of nuclear thermal propulsion the higher delta V of a typical opposition mission represents about a 50% mass penalty.

Conjunction profiles offer these advantages:

a. Long stay times at Mars with low to moderate delta V.

b. Less crew exposure to rigors of zero-g space travel and radiation environments.

c. Longer-duration launch and return windows

Conjunction profiles have these disadvantages:

a. Long total mission duration

b. Consecutive missions do not provide continuous presence at Mars unless crews stay for almost 2 synodic periods, e.g. 1200 days.

Mars Direct - The Mars direct mission profile uses conjunction trajectories. As indicated earlier, the Earth return habitat and its propulsion stage are placed on a Mars transfer by a cryogenic propulsion system (nuclear could be used). Various renderings of the architecture use aerocapture at Mars to enter an orbit for navigation update or direct entry from the approach trajectory. The entire vehicle lands; nothing is left in Mars orbit. On Mars, an electric powerplant, usually described as nuclear, is roboticaly emplaced and started. Included in the landing stage is a propellant production facility which produces liquid methane and oxygen, using hydrogen brought from Earth, as described above. This is started robotically and gradually fills the return system which was landed empty. Before the crew departs Earth during the following mission opportunity, the return system has been filled with propellant and ready to use. The crew flies to Mars on a similar profile, bringing the main surface mission habitat. The entire crew vehicle lands near the return system and the Mars surface mission begins. In the usual rendering, a second return system travels to Mars during the crew mission window, nominally for use by the next crew on the next mission opportunity. However, this is available for backup for the current crew.

As described by its authors Zubrin and Baker, Mars direct is very efficient in terms of launch requirements. Their mass estimates, especially for the Earth return habitat, are low compared to other analysts. Mars direct was compared to other conjunction profiles by STCAEM using consistent ground rules and mass estimating relationships, as was shown in figure 2-4 earlier in Section 2.1.3, where it is called "surface rendezvous". Mars direct combines surface and transportation system functions such that direct transportation comparisons are misleading. Its effective mass efficiency is approximately equal to a nuclear thermal rocket generic conjunction profile. Mars direct obtains dual use of the habitats, i.e. for transportation and surface operations. A series of separate cargo launches is not needed. Most of the return propellant is obtained from Mars.

The STCAEM fully reusable version of Mars direct proved less effective for its intended purpose of permanent base support than a reusable NTR, NEP or SEP with a Mars-based reusable Mars excursion vehicle (MEV).

Mars direct offers these advantages:

a. Dual use (transportation and surface) of habitats.

b. Mass efficient

c. Few vehicle system developments, e.g. the advance cargo and crew missions can use the same aerobrake.

d. No Mars orbit operations needed

Mars direct has these disadvantages:

a. Requires significant robotic operations on Mars to prepare the return vehicle. (There may be ways to reduce this risk by tailoring a development strategy.)

b. Abort options are somewhat limited. In particular, a landing cannot be aborted and must land close to the return vehicle.

c. Vehicle architecture cannot readily adapt to other profiles.

Generic Opposition Options (Including Swingby) - The synchronism of the opposition profile accomplishes a Mars round trip in one opportunity window. The space vehicle makes the same number of revolutions around the Sun as the Earth, usually about 1-1/2. The trip from Earth to Mars occurs early in the Earth-Mars window, and the trip from Mars to Earth, soon after Mars arrival, occurs late in the (nearly concurrent) Mars-Earth window. Since the profile is using early and late parts of the windows to accomplish the mission in a shorter time, the required energies are substantially greater than minimum. The longer the mission stays at Mars, the greater the energy required. The optimum stay time is zero.

The profile spends time at and near Mars, with a heliocentric angular rate less than Earth. It must therefore spend compensating time closer to the Sun than Earth at a higher angular rate to make the average equal to that of Earth. Opposition profiles usually travel closer to the Sun, near the orbit of Venus, on one leg of the trip but not the other. If Venus happens to be in the vicinity during the sunward pass, the trajectory design can usually take advantage of a Venus gravity assist, making the profile of the Venus swingby type. A Venus swingby can benefit on either leg of the trajectory and occasionally both. Venus swingby alters the nature of the trajectories such that longer stay time at Mars is energetically reasonable, and a Venus swingby profile will often have an optimal stay time greater than zero.

If Venus swingby is not practical (usually because Venus does not cooperate by being advantageously placed) a propulsive maneuver during the closest approach to the Sun often called deep space burn or maneuver, may be beneficial.

The energy required for this profile normally rules out all-cryogenic propulsion because the initial mass required is too great. Cryogenic/aerobraking, nuclear thermal, nuclear electric and solar electric propulsion are all candidates. Electric propulsion systems tend to be power-limited on this profile and may require longer duration than high-thrust systems.

A representative opposition profile sequence is the same as a conjunction sequence, except that a Venus swingby or deep-space burn may be interposed on one (possibly both) transfers. The differences are (a) short rather than long Mars stay, and (b) the transfer leg that passes closest to the Sun may approach a year in duration, whereas conjunction transfers, unless the very lowest energies are used, normally require six to eight months.

The wait time in Mars orbit is not enough to get much help from oblateness perturbations. If an elliptic orbit is chosen, selection of a low-energy profile must include consideration of alignment losses on Mars orbit arrival and departure. The influence of these losses can be enough to alter the choice of interplanetary trajectory. A number of choices is usually available, e.g. outbound Venus swingby versus inbound deep-space burn. They do not continuously blend into one another as is the case for conjunction profiles. Because of higher energies and alignment problems, the Mars capture and departure delta Vs for an opposition profile usually range between 2000 and 4000 m/sec. Even with the alignment problems, elliptic Mars orbits usually offer less delta V than circular orbits.

Opposition profiles offer these advantages:

a. Shorter total mission duration, by about a year.
b. Synchronism permits continuous presence at Mars, with each mission operating in a crew exchange mode.

Opposition profiles have these disadvantages:

a. Constrained stay time at Mars except in crew exchange mode.
b. Significantly higher mission energy
c. Nearly continuous (except for the short Mars stay) exposure of crews to the zero-g and space radiation environments, for more than a year. (Artificial-g space vehicles could be used.)
d. Short launch windows at Earth.

Flyby/Dash and Related Cyclers - The notion of operating an opposition profile in a crew exchange mode leads naturally to the flyby/dash and cycler profile concepts. The idea is that if optimum stay time is short and the needed stay time, i.e. for crew exchange, is also short, one can design a profile in which the interplanetary vehicle does not stop at Mars and therefore requires much less delta V. (A Mars flyby with return to Earth will often need some delta V at Mars.) Two concepts have been published. First, a conventional opposition profile without Mars stopover can be used. Such Mars flyby trajectories were published as early as the early 1960s. The Mars excursion vehicle, with the crew, separates some time before Mars arrival and "dashes" ahead, arriving Mars a few days in advance of the interplanetary vehicle. It lands, exchanges the crew, and lifts off Mars at the right time for a hyperbolic rendezvous with the interplanetary vehicle (it is on a hyperbolic, i.e. uncaptured, path relative to Mars).

If the interplanetary vehicle uses electric propulsion, it may slow down on approach to Mars and speed up again after the hyperbolic rendezvous; this reduces the delta V required of the ascent vehicle. Or it may use a gravity assisted capture in which the Mars excursion vehicle lands at the time of the assist encounter and ascends after capture is complete a few weeks later. (High thrust captures are always gravity-assisted.)

The cycler profile adds another feature: the interplanetary trajectory is shaped and controlled such that gravity assists at Earth and Mars cause the trajectory to "repeat" every Earth-Mars synodic period. The interplanetary vehicle, once placed on this trajectory, needs no further propulsion to continue repeating Earth and Mars encounters. Unfortunately, because of the eccentricity of Mars' orbit, the repeat pattern is somewhat irregular and gravity assists alone do not work all the time. A modest amount of low-thrust propulsion near aphelion is enough, and this could be supplied by solar electric propulsion.

The nature of the cycler trajectory is one short leg, on the order of six months, and one long leg, about 20 months. The entire trajectory repeats with the Earth-Mars synodic period which averages 26 months. The short leg can be either Earth-Mars or Mars-Earth but of course not both. As usually proposed, this scheme uses two cycling spaceships, one on each type of trajectory, so that passengers can take advantage of the short-leg trip time each way. "Small" taxi space vehicles are used to accomplish crew embarking and debarking from the cyclers. These taxis perform aerocapture and entry upon arrival and have enough delta V to make hyperbolic rendezvous with the cycler for crew departure (from either planet).

A significant drawback of the cycler profile is that encounter velocities at Mars are quite high, on the order of 8 to 12 km/sec. The "small" taxis turn out to be not so small.

The dash and cycler profiles have these advantages:

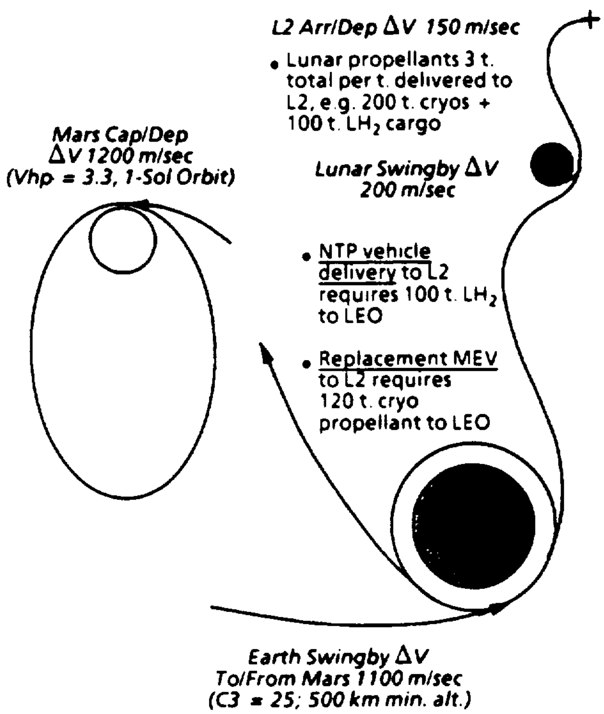
a. Less propulsion/delta V required for the large interplanetary vehicle. This becomes a very important advantage if the interplanetary vehicle must provide massive radiation shielding or other amenities that drive its mass to high values.
b. They tend to be reusable. The cycler is, and the dash profile is usually rendered that way.

The dash and cycler profiles have these disadvantages:

a. Basically suitable only for crew exchange operations, i.e. very short stay times at Mars.
b. Hyperbolic rendezvous requires precision timing of the Mars (and Earth, if required) liftoff/departure maneuver. The "pushbutton" window is short, minutes at most, and there is no second chance.
c. In the case of the cycler, encounter velocities at Mars are high.

Lunar/Asteroidal Propellants and L2 Basing - If propellants are obtained from the Moon or from near-Earth asteroids, it does not make sense to bring these propellants to low Earth orbit for use. It is more economical to use the propellants (energetically) closer to the point of production. While this might seem to dictate launching from the lunar surface, it really doesn't make sense to assemble or handle an interplanetary space vehicle on the Moon in a gravity field. Trade studies indicate that for either source, the lunar L2 libration point is the most advantageous staging base. L1 is slightly less favored energetically. Earth-Sun L1 is energetically efficient, but travel times from Earth orbit to this point are undesirably long, several weeks unless delta V penalties are accepted.

A representative mission profile is a conjunction-type with the nuclear rocket based at the L2 libration point. This is dipected in figure 2-17, with delta Vs and a representative mission sequence mass statement. The nuclear rocket is supplied with hydrogen from the lunar surface, delivered by a conventional cryogenic lunar transport/landing vehicle. The latter obtains all its propellant (hydrogen and oxygen) from the Moon. Thus no propellant is supplied from Earth. A lunar transfer vehicle system also transports Mars mission crews and support screws from Earth orbit to L2 and back. Cargo bound for Mars, such as Mars excursion vehicles and surface base cargo, is similarly transported from Earth.



NTP Start	298 t.
MTV Empty	32
Hab	60
MEV	80
LH_2 Propellant	126
L2 Dep.	5.6
Lunar Swingby	7.3
Earth Coast Mass	285.1
MCC	1.0
Earth Swingby	37.2
Mars Coast Mass	246.9
MCC	2.0
Mars Capture	34.74
Dispatch MEV	80
Remaining Mars Orbit	130.16
Dep. Mars	18.5
MCC	1.0
Earth Coast Mass	110.7
Earth Swingby	14.6
Lunar Coast	96.1
Lunar Swingby	2.4
L2 Arrival	1.7
Empty Mass	92

Figure 2-17. Mission Mode: NTP Reusable, L2 Base, Expendable MEV

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It is also possible to think of electric propulsion systems operating from an L2 staging base. One of the STCAEM trades examined this question. It was concluded that an L2 staging point was preferable to bringing the entire interplanetary vehicle to low Earth orbit by means of a lengthy spiral maneuver. The STCAEM solution used a smaller cargo-type electric propulsion system to resupply the interplanetary vehicle with propellant and Mars cargo, and a conventional cryogenic propulsion system to transport crews. It is possible to employ cryogenics for all resupply from low Earth orbit while maintaining reasonable initial mass in Earth orbit, as suggested in figure 2-18.

L2 basing has the following advantages:

a. Avoids reusable nuclear reactors being parked in low Earth orbit. (Note that the L2 point is not a stable orbit. An object "cast adrift" at L2 will probably eventually impact either the Moon or the Earth.)

b. For low thrust propulsion systems, avoids lengthy spiral flights out of and into Earth's deep gravity well. This is important because (1) solar electric systems will suffer damage to the solar arrays due to van Allen belt passage; and (2) nuclear electric systems will accumulate about twice as much run time per mission compared to the L2 basing case.

c. Energetically and mass efficient for use of lunar or asteroidal propellants.

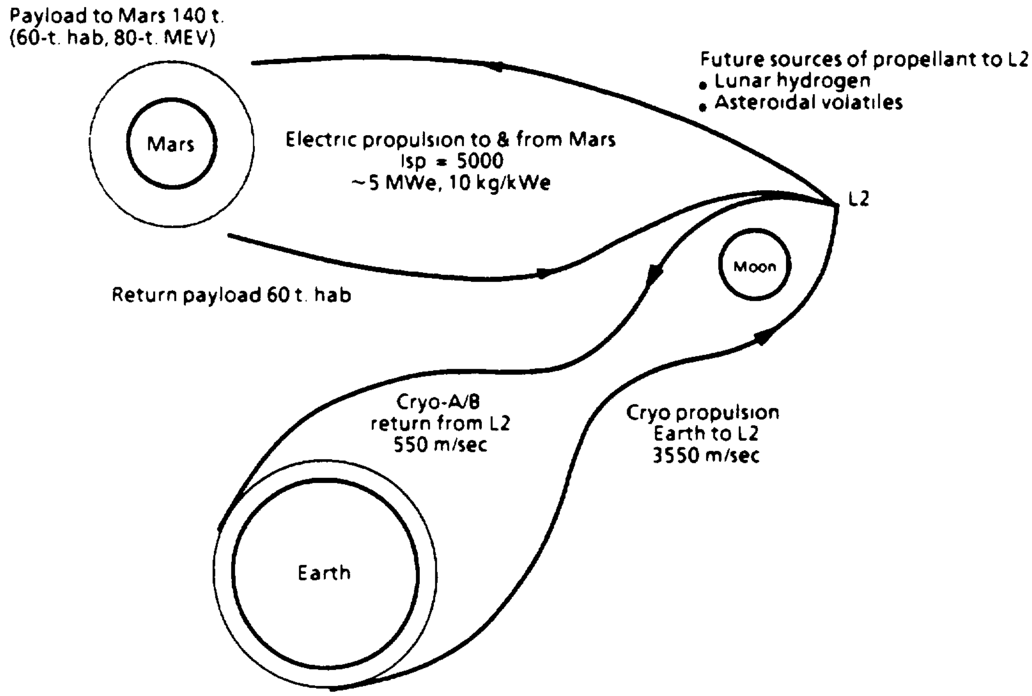

Figure 2-18. Electric Propulsion, L2 Node

L2 basing has the following disadvantages:

a. Expected to be somewhat more costly to attain initial mission capability.

b. More costly to transport people to and from the L2 point than to LEO. If significant numbers of maintenance personnel are needed to prepare and maintain a Mars space vehicle between missions, this could be an important disadvantage.

c. Launch window constraints exist for high-thrust missions using Moon and Earth gravity assists for trans-Mars injection and Earth return arrival.

2.4 GENERAL ARCHITECTURE REVIEW AND EVALUATION

As stated in the introduction of this report, the current phase of the STCAEM study was also tasked to perform a general review and evaluation of Mars architectures. This is provided in this section. An important caveat is that rationales and mission descriptions for Mars are currently unsettled (see Section 2.6 below). Architecture evaluation criteria, which must be derived from these, are equally unsettled. Consequently, a review and evaluation may at best be provisional and at worst misleading.

2.4.1 Architectures Comparison

The Inherent Difficulty of Mars Travel - Mars transportation and surface system architectures must deal successfully with the great and variable (compared to historical human missions) distance between Mars and Earth and the resulting higher performance requirements and long trip times. One consequence of long trip times is that human habitats must provide adequate accommodations for the duration. Consideration of human environmental needs and consumables derives the habitat mass to roughly 10 t. per person (this is like a space station) compared with 1 to 2 t. per person typical of the Apollo, Lunar Module, and Shuttle cabins. The long duration and complexity of the mission indicates a minimum crew of at least six and possibly eight. Habitat systems for Mars transfer are estimated as 60 to 80 t. mass. The habitat mass for a 180-day one-way transfer might be reduced to about 40 t. (six people).

The gravity well of Mars, in delta V terms, is about twice that of the Moon, and about half that of Earth (the geopotentials differ by about factors of 10). Although aerobraking can be used for Mars landing, Mars excursion vehicles approach twice the mass of lunar ones; a typical gross mass value is 80 t. If an Earth entry vehicle is used for crew return to Earth at the end of the mission, its mass would be 6 to 8 t. This is a short-duration vehicle similar to Apollo command vehicle that serves to carry the crew through the re-entry environment to safe landing on Earth.

Each Mars opportunity is different because Mars' orbit is significantly eccentric and not in the same plane as that of Earth. The positions of the planets do not repeat exactly, even over the approximate 17-year Earth/Mars synodic "cycle". However, approximate values for delta Vs can be prescribed, such as presented in Table 2-1; while they are not accurate enough for mission design they serve for rough comparisons of propulsion systems.

Table 2-1. Representative Delta Vs for Mars Round Trip Missions

	Earth Depart	Mars* Arrive	Mars Depart	Deep-Space Maneuver	Earth to Low Orbit Arrive
Conjunction low energy	4100	1200	1200	N/R	4100
Conjunction high energy (hard year)	4300	2400	2400	N/R	4300
Opposition easy year	4300	2600	2600	N/R	4300
Opposition hard year	4600	3000	2000	2000	4600

* Assumes 24-hour elliptic orbit at Mars

Notes: (1) Opposition/swingby mission delta Vs are similar to those stated here for Opposition easy year.
(2) Electric propulsion delta Vs are highly dependent on trip times.

As noted in the Figure above, the delta V for electric propulsion systems is highly dependent on trip time. Figure 2-19 illustrates a typical dependence for a one-way Mars transfer which departs Earth near the optimum departure time. The derivation of this curve is discussed in more detail in Section 6 of this report.

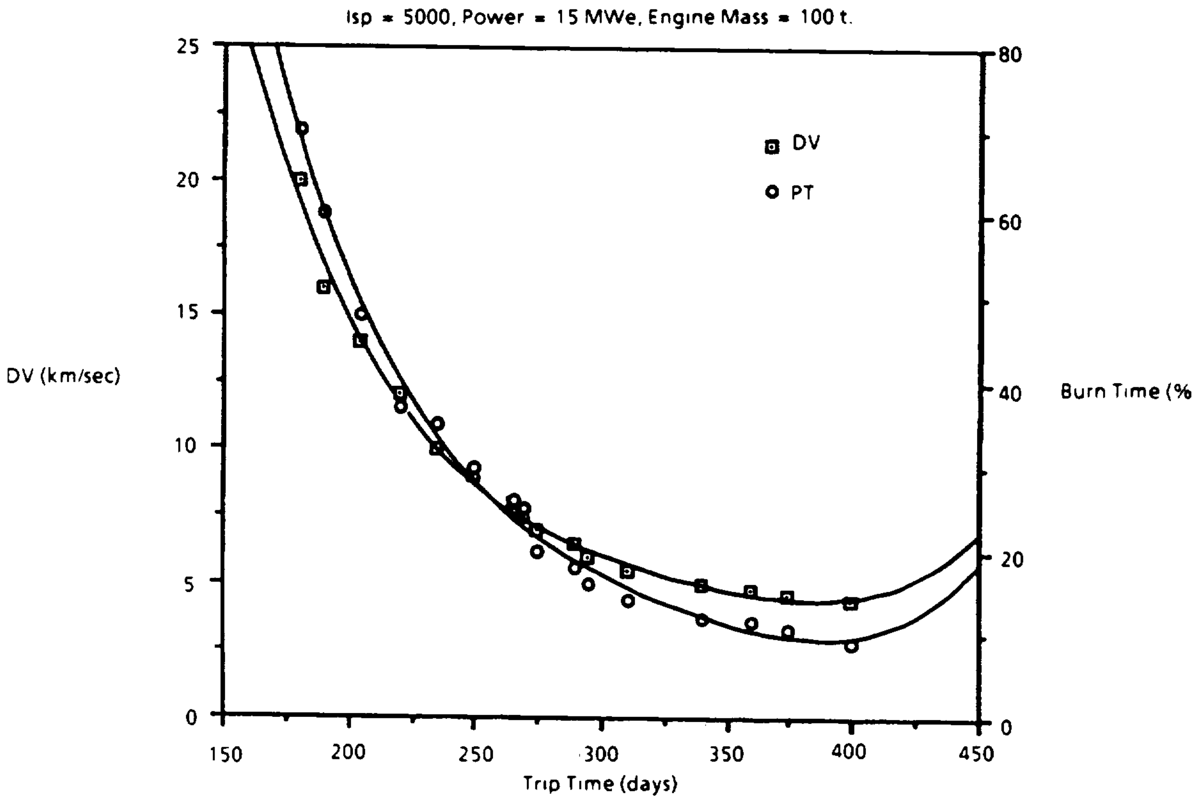

Figure 2-19. Typical Dependence of Low-Thrust Delta V on Trip Time ACS141

Table 2-2 provides a comparison of architectures in terms of propulsion mechanization. This is different than the five mission profile groups defined above, since a propulsion architecture is usually capable of carrying out more than one mission profile. These seven propulsion mechanizations are different than the original STCAEM seven as follows: Cryo all-propulsive and NTP are grouped together. Cryo aerobraking is the same as the original. NEP and SEP are grouped together. The original STCAEM version of Mars direct is replaced by the Zubrin/Baker concept of Mars direct. The ExPO reference was not represented in the original seven. The L2/lunar oxygen cryogenic scheme was determined not economically attractive and was dropped. The L2-based NTP is new. There is a related cryogenic option (not evaluated) in which both hydrogen and oxygen are obtained from the Moon. It does not have the economic drawbacks of oxygen-only, but requires roughly twice the lunar propellant production rates as the NTP option. The electric L2-based option is new.

Table 2-2. Architecture Options Summery, Characteristics in Terms of How They Accomplish Functions

Option	Exit Earth Gravity Well	Transfer to Mars	Capture at Mars	Land on Mars	Ascend from Mars	Depart Mars	Transfer to Earth	Enter Earth Gravity Well	Return crew to Earth
Nominal all-up cryo or NTP crew mission	Combined high-thrust maneuver(s)		Propulsive	MEV	MEV ascent stage	Combined high-thrust maneuver(s)		Ballistic	Crew return vehicle
Nominal all-up cryo A/B mission	Combined high-thrust maneuver(s)		Aerobraking	MEV	MEV ascent stage	Combined high-thrust maneuver(s)		Ballistic	Crew return vehicle
Nominal all-up NEP or SEP crew mission	Propulsive spiral	Low-thrust transfer	Propulsive spiral	MEV	MEV ascent stage	Propulsive spiral	Low-thrust transfer	Propulsive spiral	Crew return vehicle
Mars Direct	Combined high-thrust maneuver(s)		Aerobraking capture and descent/entire vehicle		Prepositioned Earth return vehicle fueled from Mars propellant			Ballistic	Crew return vehicle
ExPO reference	Combined high-thrust maneuver(s)		Aerobraking capture and descent/entire vehicle		Prepos. ascent stage, Mars propellant	Prepositioned Earth return vehicle		Ballistic	Crew return vehicle
NTP, L2-based	Crew & cargo, lunar transportation system	High thrust	Propulsive	MEV	MEV ascent stage	Combined high-thrust maneuver(s)		Crew return by lunar transportation system	
Electric, L2-based	Crew a& cargo, lunar transportation system	Low-thrust transfer	Propulsive	MEV	MEV ascent stage	Propulsive spiral	Low-thrust transfer	Crew return by lunar transportation system	

To obtain a complete mission architecture, one must combine a mission profile, a compatible surface mission definition, a means of delivering the surface cargo needed for the surface mission, and one or more propulsion mechanizations for the cargo and crew missions. In some cases, such as Mars direct, there is little distinction between mission profile and propulsion mechanization categories, i. e. they become a matched pair. In other cases, several propulsion mechanizations are applicable to a particular mission profile. In the present evaluation, certain limitations were placed on all possible combinations to limit the scope of the evaluation:

a. Only conjunction profiles were considered, in keeping with the current ExPO mission architecture study. Opposition profiles would in some cases alter evaluation results.

b. The surface cargo requirement of 150 t. landed before the first human mission to Mars, identified by the ExPO study, was accepted without critique.

c. It was assumed that cargo transportation to Mars would be accommodated by the same propulsion technology selected for the crew system. Accordingly, cargo transportation issues were not included in the evaluation.

d. The evaluation was conducted by judgmental scoring based on available information on the architectures. Calculations of architecture performance characteristics were done in only a few instances.

Table 2-3 compares several propulsion mechanizations according to qualitative features, benefits and issues. This complements the advantages and disadvantages of the mission profiles presented earlier. It is clear that all have good and bad features. The STCAEM assessment is that any of these mechanizations could be the favorite under plausible future determinations of Mars program rationale, mission strategy and technology status.

Table 2-3. Architecture Options Summary

Note: All options take advantage of cargo pre-placement

Option	Features	Benefits	Issues
Nominal all-up NTP crew mission	• All-propulsive • Maximum abort menu • "Sortie" MEV	• Safety potential • Insensitive to profile difficulty	• NTP ground testing • Reactor disposal
Nominal all-up cryo A/B mission	• Cryogenics + aerobraking • "Sortie" MEV	• No reliance on nuclear propulsion • Opposition capable	• Aerobrake risks • Profile sensitivity
Mars direct	• Conjunction/split • Prop/aerobraking • Direct landing • In-situ propellants	• No Mars orbit ops • Minimum Earth orbit Ops	• Reliance on robotics • Low IMLEO claim hinges on light return hab.
ExPO reference	• Conjunction/split • Nuc & A/B options • In-situ propellants • Split "sortie" MEV	• No Earth orbit ops • Reduced IMLEO	• Reliance on robotics • Lack of aborts •
Electric (NEP or SEP)	• All-propulsive, low thrust • "Sortie" MEV	• Minimum Earth obit ops • Reduced IMLEO	• EP cost • EP reliability

Selection criteria are clearly dependent on program rationale and mission strategy. To illustrate the point, table 2-4 includes selection criteria consistent with the Synthesis report selection of nuclear propulsion in 1991, and adds other criteria that appear to have equal if not greater relevance today (summer 1993).

Table 2-4. Old and New Criteria for Architecture Evaluation

		Old	New
(1)	Evolution to low recurring cost and large passenger capacity, responding to settlement as a long-range vision. Even a major human scientific exploration of Mars needs this;		X
(2)	Low or at least reasonable cost to first mission. Highest leverages: (a) Modest-size commercial launch vehicles; (b) Commonality of hardware, minimum no. of development projects	X	X
(3)	Commonality of overall architecture with other space activities. • Future evolution of space transportation and operations architecture should accept exploration as one of its missions; • Exploration program should accept the constraint of compatibility with a reasonable overall architecture.		X
(4)	Multiple-use technology developments, i.e. benefiting society on Earth, to make the near-term economic benefit of exploration important in its own right;		X
(5)	Safety, because such a mission is risky at best; and	X	X
(6)	Acceptable development and operational risk consistent with themes 1 through 4.	X	X
(7)	Minimum mass in low Earth orbit (often used but not by STCAEM)	X	

Table 2-5 shows scoring of the propulsion mechanizations for the two sets of criteria, and figure 2-20 illustrates the total scores in bar chart format. That the scores are overall greater with the greater number of criteria is simply due to more criteria leading to higher scores. The importance of the Figure is that under criteria seen as important during the Synthesis activity, nuclear thermal propulsion scores highest, whereas when newer criteria are added, other options score higher. This serves to underline the earlier assertion that selection of preferred architectures is sensitive to program goals and rationales. Until these are resolved it is premature to select a "best" architecture.

Table 2-5. Provisional Evaluation

Criteria

Architecture (Score #1) [Score #2]	Evolution to Low Rec. Cost	*Low Cost to 1st Mission*	Common w/General Architecture	Multi-Use Technology	*Safety*	*Dev/Ops Risk*
Cryo A/B (6) [6]	No (0)	Unclear (3)	No (0)	No (0)	Dubious (2)	High (1)
Synthesis NTP (13) [15]	Better than Cryo A/B (2)	Better than Cryo A/B (4)	Some (1)	No (0)	Good (5)	Moderate (3)
Zubrin Mars Direct (8) [14]	Potential (3)	Depends on Robotics (4)	No (0)	Some (3)	OK (3)	High (1)
ExPO Mars Surface Rendezvous (2) [7]	Not Clear (2)	Dubious; too many develop-ments (2)	No (0)	Some (3)	Poor (0)	Higher (0)
NTR/L2 (Lun/AST H_2) (5) [20]	Yes (5)	Unclear (2)	Possible (3)	Yes (4)	Good (5)	High (1)
NEP/SEP; LEP (8) [21]	Yes (5)	Potential (1)	Yes (5)	Yes (5)	SEP/good NEP/Maybe (3)	Moderate (2)

2.5 TECHNOLOGY ASSESSMENT

Technology assessments have been done frequently during the various studies of exploration missions since 1988. Most of these have strived to create an "absolute" technology list. In fact, priorities for some technologies are architecture-dependent and priorities for others may be sensitive to the amount of technology funding that may be made available, and to technology developments embedded in current or planned development activities for other missions. These points are discussed in what follows.

2.5.1 Technology Candidates and Groups

The technology candidates were organized into four groups: Life support; propulsion and power; aerobraking, and testability.

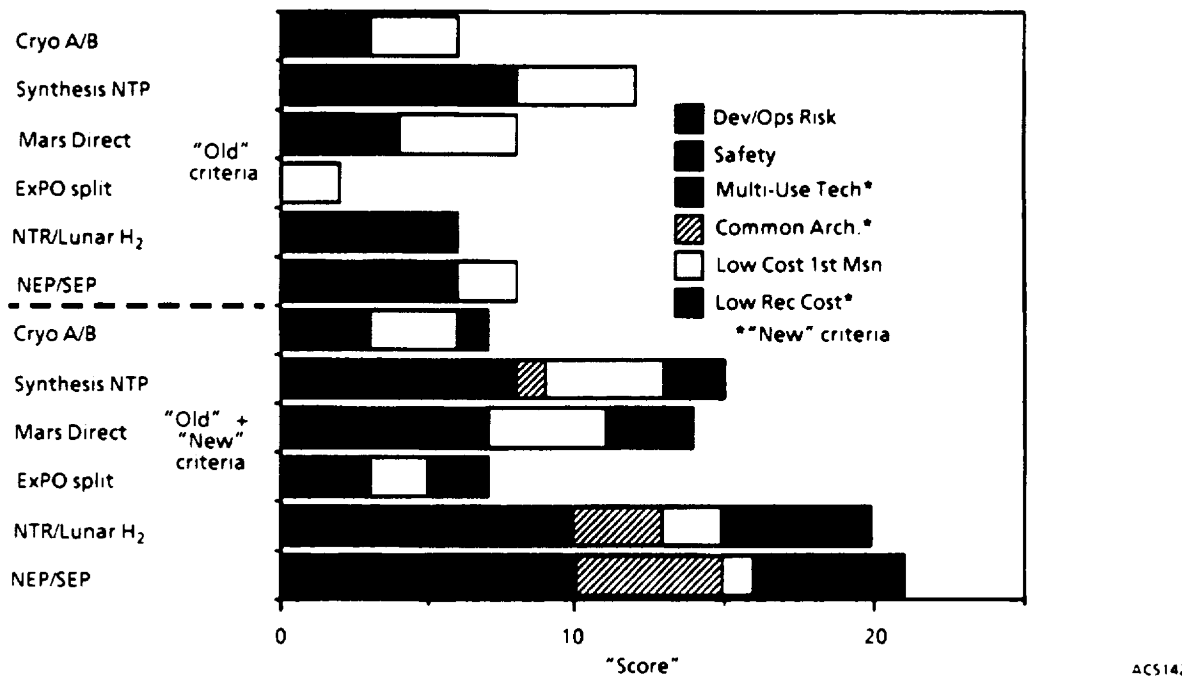

Figure 2-20. Scored Evaluation re Long-Term Utility for Mars Exploration and Settlement

2.5.2 Technology considerations

Relative importance of technologies were assessed based on the following considerations:

a. Contributions to mission safety through enabling of architectures or mission profiles that offer greater safety by virtue of fewer critical events and/or more accessible abort modes.

b. Benefits through reduced use cost, typically because of greater reuse of space hardware or reduced mass of Earth launch to conduct a mission.

c. Technology advancement cost, which must be evaluated relative to reduced use cost.

d. Readiness timing, in terms of the present state of the technology versus the state needed prior to a development program, the timing of the need in the overall program framework, and the time needed to reach an adequate readiness level.

2.5.3 Technology Attributes

We concluded that technologies should be evaluated in terms of four attribute categories (each technology may have a characteristic in more than one category):

a. The current technology is "woefully inadequate" or nonexistent: something must be done to bring the technology to the level required for application.

b. Need for the technology is architecture-driven; that is, required by some architectures but not others. In this case, the "something" that "must be done" can be a change of architecture.

c. The case for advancing the technology is basically a cost/benefit matter, i.e. its priority is driven by investment versus payoff.

d. The technology has a very long lead time; early research-oriented investment is needed to to define the advancement path. In these cases, the eventual value of the technology may also be in question and early research can clarify this point. Often, significant progress can be made in these technologies at relatively low funding, and a real success might have major payoff. These are therefore prime candidates for investment if the available budget is small, because a significant return might be realized, where a technology beyond the research stage may require substantial funding to make any progress at all.

In addition, it is important to understand to what degree ongoing programs may be advancing the technology, as in the case of Space Station and physico-chemical life support.

2.5.4 Evaluation: Timing of Needs and Priorities

Table 2-6 presents a summary of the evaluations for each technology area. Most of these are self-explanatory but the items on nuclear power and propulsion are controversial and need special discussion.

Nuclear Power and Propulsion - This area comprises nuclear thermal propulsion, nuclear electric propulsion, nuclear electric power for other space uses, especially planet surface power, and in the future, more advanced means of extracting useful propulsion or power from the energy of nuclear reactions.

Nuclear technology is the target of much criticism because of association with nuclear weaponry, nuclear powerplant accidents (which, even including Chernobyl, have been relatively benign compared to the consequences of mining and burning coal), high costs including severe cost overruns, worrisome and dangerous byproducts such as very long-lived nuclear wastes, the highly arcane nature of the technology itself, and an attitude occasionally exhibited by nuclear technologists of "it's too complicated for others to understand; we know best how to make it work; trust us". Not surprisingly, there is a substantial public and political distrust of anything nuclear. This extends to nuclear power and propulsion for space applications. The recently revealed fact that the Air Force was developing a nuclear rocket in great secrecy does not make the situation any better. To the present day, a clear and concise need for nuclear technology in space has arisen only for RTGs; these have been used on several missions. Low and modest-

Table 2-6.

Nuclear Thermal Propulsion

Safety:
- Crew and public safety of paramount importance but not clear there is a technology issue.
- Selection of reactor disposal scheme may identify safety technology requirement.

Use Costs:
- Significant use cost benefit identified over cryo propulsion and cryo/aerobraking.
- Benefit justified advancement and development cost in most Mars scenarios.

Technology Advancement Cost:
- Major technology advancement budget item, ~ $2 billion over 6 to 10 years.
- Fuels technology advancements (first step of advancement program) has modest cost but requires new or upgraded test reactor facilities.
- STCAEM trades recommended carbide fuels technology, >900 Isp.

Readiness Timing:
First Mars cargo mission for all-up test of system. Lunar rehearsal test provides added risk reduction; advances need date ~2 years.

Categories:
- Current technology (NERVA) mothballed; far from flight ready.
- Architecture-driven.
- Payoff justifies investment re cryo or cryo/aerobraking, but nuclear or solar electric can be cost competitive alternatives.
- Lead time ~ 12 years to first flight use; somewhat technology dependent. NERVA technology ~ 8-10 years; very advanced ~ 15 years.

Nuclear Electric Propulsion

Safety:
- Crew safety and continuous nature of electric propulsion drives need for long life and high reliability and redundancy management.
- Selection of reactor disposal scheme may identify added technology requirements.

Use Costs:
- NEP potential for low use costs; reusable profiles and low propellant resupply.
- Concepts for expendable NEP profiles defeat the advantages of NEP.
- NEP cost effectiveness potential is very sensitive to trip time requirements.

Technology Advancement Cost:
- Major technology advancement budget item, >$2 billion over 10 - 12 years.
- Commonality potential with planet surface power should be fully exploited.
- STCAEM trades recommended modular potassium Rankine systems.

Readiness Timing:
- First Mars cargo mission for all-up test of system; should return to Earth for diagnostics and reuse.

Categories:
- Current technology "woefully inadequate".
- Architecture-driven but must be considered jointly with planet surface power needs.
- Payoff justifies investment re cryo or cryo aerobraking. Comparison to NTP must include planet surface commonality.
- Lead time >12 years to first flight use; life testing required; qualification life may be as great as 25,000 hours.

Solar Electric Propulsion

Safety:
- Technology advancement must demonstrate redundancy management method that capitalizes on great inherent redundancy of solar electric system.

Use Costs:
- SEP potential for low use costs; reusable profiles and low propellant supply.
- SEP cost effectiveness potential is very sensitive to trip time requirements.
- STCAEM trades showed that array production cost must be below ~$200/watt for SEP to deliver economic benefit.

Technology Advancement Cost:
Issue is demonstration of low production cost for high-performance lightweight arrays. No estimates are available.

Readiness Timing:
First Mars cargo mission for all-up test of system; no estimates of advancement time required; Believed to be about 8 years including production cost demonstration.

Categories:
- Current array performance/mass within factor of 2 of reasonable targets.
- Most aspects of production technology probably exist in commercial industry.
- Architecture-dependent. Planet surface power commonality is questionable.
- Payoff justifies investment ***if and only if array production cost is low.***
- Lead time 8-10 years to first flight use.

Special Note:

SEP is not suitable for outer solar system use.

Table 2-6. (Continued)

EVA (Suits)

Safety:
- Overriding requirement in suit design
- Safe operation must be preserved in daily routine use and over long life.

Use Costs:
- EVA systems absolutely essential for mission; need for the technology is not just a matter of in-use cost.
- Benefit justified advancement and development cost in most Mars scenarios.

Technology Advancement Cost:
- EVA equipment technology advancement can be moderate in cost.
- Operational equipment development is expensive because of the strict safety requirements.
- Essential to ensure that technology base is adequate and fit for use during advancement phase to avoid full scale development of inadequate equipment.

Readiness Timing:
- Operational systems for initial Mars exploration phase.

Categories:
- Current technology "woefully inadequate".
- In-use payoff for high durability and long-life systems.
- Not architecture-driven.
- Fully adequate operational systems only after field experience. Technology advancement should not strive for what is only achievable with field experience.

Cryogenic Fluids Management

Safety:
- Inadequate technology leads to safety issues including fire and explosion and crews being stranded in space due to loss of propellant.

Use Costs:
- This technology is essential for nuclear thermal and cryogenic propulsion.
- Cryo propulsion needed in landers even with alternative transfer propulsion.
- Alternatives are costly by comparison.

Technology Advancement Cost:
- Depends on specific need. Can be modest to moderately expensive.
- This area has been plagued by high-cost space experiment proposals.

Readiness Timing:
- High-performance insulation and tank pressure control for first lunar mission. Full menu of capability for Mars; specific requirements are architecture-dependent.

Categories:
- Current technology "woefully inadequate".
- Not architecture-driven (all architectures need it) but specific technology advancement plan needs to be coordinated with architecture evolution.
- Payoff justified the investment. One of the highest payoff areas identified by STCAEM trades.
- Lead time can be relatively short if incremental flight test strategy is built into the architecture evolution.

Aerobraking Thermal Protection Systems

Safety:
- TPS is crew safety critical re burnthrough.

Use Costs:
- Missions are not economically practical without use of aerobraking/TPS for planetary entry and landing.
- Cost benefits will accrue to increased technology capability for reuse after severe entry heating conditions.

Technology Advancement Cost:
- A few millions per year is adequate for materials development.
- Active TPS, e.g. transpiration cooling, requires more expensive testing.
- Inadequately developed technology can severely impact system development costs, e.g. shuttle tiles.

Readiness Timing:
Needed for initial missions; improvements can be inserted into program later.

Categories:
- Current technology is usable for all mission needs, but only for single-use heat shields in most cases.
- In part architecture-driven, e.g. aerobraking verses propulsion for Mars capture.
- Reuse payoff is architecture-dependent.
- Materials development lead time depends on the material; typically 2-4 years.

Table 2-6. (Continued)

Aerobraking GN&C

Safety:
- Crew safety critical, hardware, software, and algorithms.

Use Costs:
- Not a use cost issue; function is required to perform mission.

Technology Advancement Cost:
- One to a few millions per year, depending on how many paths/ initiatives are undertaken.

Readiness Timing:
- Needed for first missions. Improvements can be phased in later.

Categories:
- Current technology generally adequate; marginal performance in some cases.
- Architecture-driven re specific applications.
- Major payoff for advanced techniques such as neural networks because of reductions in hardware/software costs.
- Lead time to develop new techniques 2-4 years.

Lunar Transfer and Lander Engines

Safety:
- Descent to landing and ascent are life-critical functions.
- Successful propulsion operation must be assured; redundancy is a valid solution.

Use Costs:
- Not particularly a use cost issue; function is required to accomplish mission.
- Improved engine performance leads to lower use cost. High Isp is important.

Technology Advancement Cost:
- Adequate technology advancement pace needs a few tens of millions per year.
- New engine full-scale development is at least a few hundred millions.
- Minimum acceptable program is to add throttling to RL-10 derivative; development cost more or less 10% of new engine.

Readiness Timing:
- Minimum capabilities needed for first lunar mission.
- New engine could be phased in later if desired.

Categories:
- Current technology is usable with throttling and with testability (see later sheet).
- Not architecture-driven.
- Moderate favorable cost/benefit ration for new, high-performance engine.
- Lead time 2-4 years technology, 6-8 years for engine development.

Special Note:

A new storable propellant or methane engine may be needed for Mars ascent. Technology advancement is needed in the case of advanced storables.

Bioregenerative Life Support

Safety:
- Not applicable to transfer vehicles.
- Benefits on Mars can be obtained through food reserves.
- If bioregen also used for air and water, adequate backups must be provided.

Use Costs:
- Large long-range benefit for permanently occupied Mars sites; can reduce life support consumables by more than half

Technology Advancement Cost:
- Near term is research-oriented, modest cost, to establish basic science.
- Integrated systems expected to be typical of life support system costs.
- Opportunity should be taken to obtain in-use experience on space station where bioregen can "pay its way"

Readiness Timing:
- Prototype systems for Mars exploration phase.
- Operational systems for Mars permance phase.

Categories:
- Physico-chemical is adequate until settlement.
- In-use payoff for Mars surface as soon as serviceable systems are available.
- Not architecture-driven.
- Very long lead to operational systems; research and development justified to determine realistic technology advancement and development paths and schedules

Table 2-6. (Continued)

In-Situ Propellants and Fluids

Safety:
- Not a primary technology issue. Crew safety sets requirements on process reliability as source of life-critical propellant and how used in architecture.

Use Costs:
- Very favorable when properly inserted into architecture, assuming current estimates of performance/mass characteristics are valid.
- Very sensitive to ratio of annual production/installation mass; needs to approach 1 or better (except for hydrogen).

Technology Advancement Cost:
- Early process research inexpensive; $1 million/year recommended.
- Artemis-class experiments on Moon are few millions.
- Experimental prototypes during early human lunar mission ~ $100 million.
- Oxygen/methane form Mars atmosphere may not need technology experiments on Mars.

Readiness Timing:
- Lunar propellant ~ 50 t./year in time for permanent occupancy phase.
- Mars propellant timing architecture-dependent.

Categories:
- No technology (beyond laboratory bench-top experiments) presently exists.
- Some of the Mars methane processes have been used in terrestrial industry.
- Architecture-driven.
- High payoff
- Lead time requirements presently not well-understood.

In-Situ Structures

Safety:
- It is frequently not recognized that the most useful in-situ derived structures will have to safely contain ~ 1 atm. pressure.
- Will be subject to the usual stringent safety requirements, e.g., like aircraft cabin.

Use Costs:
- Not much benefit before a settlement or industrialization phase.
- Settlement or industrialization probably cannot be affordable without this technology.

Technology Advancement Cost:
- Unknown.
- Some ongoing university research; should be expanded.

Readiness Timing:
- Unknown; probably not less than 20 years.

Categories:
- No technology exists beyond a few very small experiments with analogous materials.
- Program goal driven; not architecture-driven.
- Very high payoff if and when needed.
- Lead time unknown.
- Appropriate to expand university research to better define the technical possibilities and problems

Testability

Safety:
- Crew safety critical.
- Issue is safe and reliable operation of systems after long periods in space, and how to provide testability that assures very high confidence systems will work when called upon.

Use Costs:
- Not a use cost issue, except that not having this technology could drive us to high-cost mission designs and operational practices to ensure adequate safety.

Technology Advancement Cost:
- Unknown; specific technology requirements not identified (even the functional requirement is incompletely known).

Readiness Timing:
- Needed for first missions.
- Mars requirements is more difficult than lunar because of longer times and greater difficulty of rescue

Categories:
- Current technology "woefully inadequate".
- Artificial intelligence won't solve this one. It does not appear to be an invention problem. The issue how to assure that a system which is not operating will when called upon.
- Not architecture-driven.
- Payoff appears to be high
- Lead time unknown. The specific technology requirement needs to be defined.

Table 2-6. (Continued)

Physico-Chemical Life Support

Safety:
- Adequate durability for long-duration missions is essential.
- Mature hardware more important than latest highest-performance gadgets.

Use Costs:
- Need closure on water and oxygen.

Technology Advancement Cost:
- Space Station contribution not clear.
- Russian contribution not clear.
- If hardware maturity and durability must be accomplished through technology and development programs it will be expensive and probably inadequate.

Readiness Timing:
- In time for first Mars mission.

Categories:
- Inadequate; durability presently is not adequate for a Mars mission.
- Not architecture-driven.
- Open life support would cost over $1 billion more per mission just for launch.
- Lead time is ~10 years of in-use testing but space station use counts.

Aerobraking Structures

Safety:
- Crew safety critical re structural failure.

Use Costs:
- Reduced structural mass has moderate to high leverage on use costs.
- Deployable or robotically assembled structures may have great leverage on space operations cost.

Technology Advancement Cost:
- Highly application-dependent.
- Generally in the few millions to few tens of millions per year.

Readiness Timing:
- Needed for first missions; improvements can be phased in later.

Categories:
- Current technology is adequate.
- In part architecture-driven; e.g., aerobraking versus propulsion for Mars capture.
- Payoff for higher-performance structures is almost universally very good.
- Materials lead times are notoriously long, up to decades.
- Design application lead times are short, 2-3 years or less.

level nuclear technology programs have existed in the DoE and NASA for many years based on a general perception that they will be needed "someday". In the past year, political tolerance and support for these activities based on potential future need has apparently evaporated. The programs are being shut down, apparently based on piecemeal decisions made in the vacuum of lack of clear policy.

This is all to make the point that nuclear technology is not a question of priority, it is a question of national policy. Space technology has become politicized, at least to the point where advocacy by expert technologists because something is "good technology" or because of potential future need, is not enough. Apollo lifted off for the Moon on engines funded by the Air Force for four years before Apollo started because "we ought to have a million-pound-thrust engine". Today that doesn't work. The U. S. needs a national policy on the development and application of nuclear technology in space. No policy is a policy to shut down existing programs and not develop space nuclear technologies.

Development of nuclear technology policy for space applications should recognize several considerations:

a. While there are alternatives for many of the proposed future applications of nuclear space power or propulsion, they will generally cost more and take longer; that is why nuclear technology was recommended for these in the first place.

b. Nuclear technology development needs continuous long-term support to maintain the research and test facilities and to develop and maintain the needed human expertise and analytical tools.

c. Nuclear space systems need to be designed for long life and high reliability, and the technology needs to support this. Mission studies often push the technology to performance limits to minimize mass, but this does not minimize cost or maximize safety. Most studies have given little consideration to how end-of-life nuclear systems will be dealt with, or what it will cost. The utility of nuclear technology in space, in the broadest sense, is enhanced by long life; we should think in terms of system design life of 20 to 100 years.

d. Nuclear technology is essential to the long-range future of space exploration and development. Humankind will probably never leave the inner solar system without some practical form of advanced nuclear propulsion.

Testability - Space systems have a serious reliability and testability deficiency. Commercial airlines are held up to public criticism if they do not achieve an on-time dispatch reliability of about 95%. Modern jetliners are comparable in complexity (parts count, etc.) to space vehicles. The on-time dispatch reliability of the space shuttle is poor, much less than 50%. Unless one drives an old clunker, the on-time dispatch reliability of a personal automobile is on the order of 99.9% Autos are of course less complex than space vehicles, but how much less? Also, the cost of auto hardware (per pound) is less than 0.1% that for typical space vehicles.

Many shuttle launch delay problems are caused by failures or instrumentation problems in hardware that cannot be functionally tested until the last few seconds of countdown, when subsystems are started up for flight. If these items could be adequately tested during launch preparations, fewer launch delays would occur because failures could be detected and corrected in advance.

It is not practical to consider using a hardware/software/test/ operations technology that achieves 10% dispatch reliability for exploration-class missions. Too many things must work to get a mission launched and to achieve mission success and safe return of a crew. If a launch from Mars encounters hardware failures, one can't obtain replacement parts by scavenging them from another vehicle. Improvements must be made:

a. Higher hardware/software reliability through maturity and through design for reliability instead of simply relying on redundancy for safety.
b. Improvements in design and advancements in technology to enable adequate testing of dormant systems (such as a Mars ascent propulsion system that will not be called upon to operate until ascent from Mars). Abort and rescue mission options depend on timely knowledge that a critical system is non-operable.
c. Improvements in quality practices to minimize the chance of failure due to faulty design or workmanship.

Finally, on testability a warning: Artificial intelligence is not a panacea and may be of no help at all. The technology of automated testing and diagnostics for systems which are operating is far advanced. Artificial intelligence is sometimes used, but ordinary software also works well. The issue is to adequately test a system which is not operating, and artificial intelligence offers no help here. It will be necessary to get into the inner workings of each critical subsystem and ascertain, test point by test point:
a. Is this item testable? How?
b. If not, can it be redesigned to make it more testable? (For example, a critical valve could be designed to go through a test cycle which determines that the actuators and sensors are working and the valve is not "stuck", without opening the valve and permitting flow.)
c. If not, can the system be redesigned to eliminate the item or make it testable?
d. What are the specific technology advancements and advanced developments needed to demonstrate testability?

2.5.5 Funding Constraints, Technical Capabilities, and Projected Readiness

Funding currently available for exploration-related technology development is not enough to take on major technology advancement projects. The best strategy is to use the modest resources available to make advancements in research-level technologies. Three categories are recommended: bioregenerative life support (some research is already going on in this area), *in-situ* materials processing, and testability. University research is under way in *in-situ* materials and could be expanded.

In testability, new research should be focused on the issues described above, covering testability for dormant systems. It is recommended that propulsion systems be the prime subject of initial investigations, and that initial research concentrate on identifying design features and technology advancements at the propulsion system component level which could become the subject of laboratory technology advancement programs.

When additional funding becomes available, the technology areas listed earlier in table 2-6 should be considered for funding. Each will require annual funding in the low tens of millions to make reasonable progress. It is better to adequately fund one or two areas than to inadequately fund all.

While the lander engine area is ranked relatively low in priority on the premise that an adequate lander engine can be developed without a technology advancement program, consideration should be given to the state of rocket propulsion technology in the U. S. There are no significant technology advancement programs or engine developments presently under way. One beneficial approach might be to combine a program in testability with one in lander/ascent engine development

Space nuclear technology, as mentioned above, needs formulation of a national policy. Advancements in fuel form technology, for nuclear thermal propulsion reactors and nuclear electric power reactors, needs funding on the order of tens of millions annually. Power conversion machine technology needs at least $10 million annually; Brayton and potassium Rankine equipment should be funded.

To be ready for incorporation into full-scale development programs, nuclear technology needs to be carried to the level of prototype rocket reactors and engines, and prototype electric powerplants. The notion that reactors and power conversion systems need not be coupled together for integrated testing may be acceptable for low power thermoelectric conversion systems but is not appropriate to high-power dynamic thermal cycle systems. Both nuclear thermal propulsion and nuclear electric power systems will require major new or refurbished test facilities for this phase of development. The funding requirements could grow to hundreds of millions annually. The funding requirements for this phase of technology advancement is a further reason for a national policy on nuclear power and propulsion technology for space applications.

2.5.6 Summary of Technology Recommendations

a. Initiate "seed money" funding of research-oriented areas: bioregenerative life support, *in-situ* materials, and testability.

b. Fund technology advancement areas in priority order (see table 2-7) as funding becomes available. It is better to fund some areas adequately than all inadequately.

c. Develop a national policy on nuclear power and propulsion for space applications. A key part of this recommendation is to analyze future needs and alternative energy sources so that the policy can be based on potential mission needs and on the consequences of not having nuclear power and propulsion technology available when needed. NASA should take the lead in developing such policy for civil space applications, coordinating with DoD to identify common interests.

A further summary is provided in table 2-7.

Table 2-7. Technology Conclusions

National R&D Policy Issue: Nuclear Propulsion and Power
- Needed technology funding is large compared to current space R&T budget.
- Policy issue, not a priority issue. With policy in place, priority will fall out.

Technology Advancement Funding Recommended; in priority order
- EVA suits.
- Cryogenic fluids management.
- Aerobraking TPS
- Aerobraking GN&C.
- High-performance solar array production cost reduction (to $200/W @ 4 MWe/yr).
- Lander engines.

Technology Research Funding Recommended; equal priority
- Bioregenerative life support.
- *In-situ* materials: propellants and structures.

Need to Understand the Technology Requirements: Testability
- Technology for automated diagnostics of operating hardware is mainly in hand.
- How to test non-operating critical system, e.g., lander engines?
  - Not an AI question; need an integrated technical approach.
  - Anticipate the technical approach will identify component technologies.

Relationship to Current Programs
- Physico-chemical life support.
- Aerobraking structures.

2.6 RECOMMENDATIONS FOR FURTHER ANALYSIS

This is the final report for the STCAEM study. The study lasted four years, covered almost all of the period of the SEI activity initiated by President Bush, and continued for several months after the 1992 election, which effectively marked the end of SEI. Accordingly, it seems appropriate that this report make recommendations for future work which go beyond the strict boundaries of the statement of work. These recommendations are aimed at specifying the kinds of analyses that could place the exploration of Mars on the U. S. national agenda as an activity highly relevant to national needs and goals.

2.6.1 The Rationale Problem

Clearly, there was a rationale problem with SEI, as suggested by figure 2-21. President Bush's speeches on the subject did not establish rationale beyond "leadership". While leadership may be an important rationale, it is necessary to state clearly why exploration of Mars establishes, or contributes to, leadership. This was missing.

- It's not a cold war imperative like Apollo
- The Supercollider seems to be at the upper edge of politically feasible cost for a "big science" project
- Nuclear rockets and long-duration space habitats don't seem to have much to do with national competitiveness or economic growth.
- The environmental connection, although it exists, is tenuous.
- "Spinoffs" don't work or SEI would have got some funding.
- Presidential advocacy isn't enough for a lengthy program. Administrations change.
- International cooperation can be a highly beneficial implementation but it's not a rationale
- "Because it's there" is probably better than any of the above. (The leadership thing; it's the kind of thing that world leader nations do.)
- Stimulus for excellence in young people is a good one. But what turns them on is a space program they can be part of. Six people to Mars is not that.

Figure 2-21. The Rationale Problem for Human Mars Missions

NASA attempted to develop rationale after the fact, by observing that space exploration stimulates education, creates jobs, and so forth. These were in the nature of "spinoff" rationales simply by being after the fact of setting the goal by the President. It was never clear how Mars exploration would reduce the deficit, improve national competitiveness, reduce crime, solve health care, or clean up the environment. In fact, it might contribute to all these things but an effective case was not made. The goals of a space exploration program must be actually derived from national needs and goals, and widely recognized as having been so derived. Otherwise, rationales relating said space program to these needs and goals are widely regarded as fabrications even if they are substantially accurate.

It is useful to consider historical precedents for initiation of major space programs. Apollo was simply a Cold War imperative. Its purpose was to demonstrate the superiority of U. S. technology, in a field which the Soviets had promoted as the premier arena of peaceful superpower competition in the eyes of world opinion. Valiant efforts on the part of NASA to obtain funding to extend Apollo to a lunar base, even a temporary one, were not successful. Once Apollo succeeded, a lunar base was not a Cold War imperative.

Skylab was approved as a flight experiment, in a time when major flight experiments could be approved if the price was right and important technical advances were promised.

The shuttle and the space station were both supported on the basis of economic considerations. Shuttle was intended to reduce the cost of space transportation and the space station is intended to become a national laboratory in orbit, serving many lines of research and development. Both these programs maintain the U. S. civilian manned space flight institution without large annual funding peaks such as needed by Apollo.

Space science missions are supported for their scientific return and their contributions to education. It appears unlikely that a new, costly exploration program can be funded on the basis of scientific return. The Superconducting Supercollider seems to be at the upper edge of feasible cost for a "big science" program, and exploration of Mars will be much more costly than that.

Presidential advocacy also did not serve to get SEI moving.

International cooperation is a highly beneficial implementation, but by itself is not a rationale for Mars exploration. International cooperation can be a part of any space activity.

Two of the rationales offered for Mars exploration are promising. "Because it's there" is probably better than any of the above. Exploring Mars is the kind of thing that world leader nations do. A group of world leader nations can of course work together to make the endeavor truly international. Stimulus for excellence in young people is also promising. It is argued that people need a vision of a hopeful future. But that vision needs to depict a space program many people can aspire to participate in. The SEI scenarios of six people to Mars don't offer that vision.

2.6.2 Settlement as Vision and Rationale

The vision that fits better than any other is already mentioned in national space policy: "Expand human presence into the Solar System". It can begin with settlement of Mars. Development of the Moon as a resource base and industrial park is also a part of this vision, but Mars is the logical settlement goal. It is the most habitable planet in the Solar System after Earth. Not actually very habitable because of its severe environment, Mars has a reasonable day/night cycle, some atmosphere, and all of the raw materials needed to construct a civilization. Some investigators have put forth plausible schemes for "terraforming", i.e. altering Mars' environment to the point that people could live there without spacesuits. Mars is also seen a treasure trove of science including planetology, Solar System history, and paleobiology. Some scientists speculate that life may still exist on Mars in sheltered areas where energy is available, such as underground pockets of liquid water heated by low-level volcanic activity.

The main reasons for Mars settlement according to this vision are not science, but economics, the romance of a frontier, and advancement of human civilization. Dr. Thomas Paine described ten reasons for settling Mars, summarized in figure 2-22.

Economic Development - long term investment in a creative new growth economy
Limitless Growth Potential - vast untapped resources, eliminate limits to growth
National Pride and Leadership - history's greatest high-tech adventure
Religious, Ideological and Hunamistic Values ... preserve life and expand [humanity]
Martian Descendants ... growth of families, race and ethnic groups on new worlds
A Fresh Start ... civilization working toward limitless future for mankind
Technical Pilgrim's Haven ... technically oriented frontier societies
High Productivity Systems Driver ... development of reliable robotic production
Research and Exploration - Unique R&D base and technical society ... for research
Prototype Extraterrestrial Community ... for future generations to [use farther out]

Figure 2-22. Dr. Tom Paine's Reasons for Mars Settlement (Abridged)

The settlement vision sets high aspirations for the human species and asks a profound question: Are we humans a one-planet species or is it possible to expand our habitat beyond Earth, into the solar system, someday beyond?

As Apollo pictures of Earth as a "blue marble" in the blackness of space changed human perspectives about Earth, either a yes or no answer to the settlement question will cause humanity to view its relationship to the home planet differently, and design a future here on Earth for a sustainable future, while trying to create a future that reaches beyond Earth.

This vision offers a Mars settlement devoted to exploration and science on a new world which will eventually become a second home for mankind. In the process, important development in addition to science will occur. Suggestions for missions beyond science include:

a. Development of settlement construction technology
b. Development of closed environment agriculture (bioregenerative) food production technology
c. Development of closed cycle waste management technology
d. Development and maintenance of an independent agricultural gene pool (easily worth the cost of settlement if settlement is done cost-effectively)
e. Development of advanced robotics and AI technology
f. Development of terraforming science.

This kind of program could draw broad public support. There is a place in it for people, eventually a great many people.

2.6.3 Economic Factors

Historically, settlements have produced major advances in products and services, technology, science and the arts, and societal and political ideas. All have economic value as well as values beyond economics.

While the long-range economics of Mars settlement is highly uncertain and has not been analyzed, four points can be made:

a. Settlement is the least expensive way (to the sponsoring states) to maintain a permanent human presence on Mars more than a few people.
b. A sizable settlement on Mars, even if mainly scientific, might generate as many as 100 "commercial" passenger trips per Mars opportunity, representing a substantial space business opportunity.
c. One can imagine "slots" in settlement billeting being sold for enough (say $50 million) to make a profit transporting people to Mars.
d. There is, of course, no cost to Earth for anything the settlers make for themselves through self-sufficiency.

One of the main questions on settlement economics is the cost of transportation to Mars. Rough calculations indicate an eventual trip price, based on known technology, about $30 million per round trip, including cost of money. This calculation is summarized as follows:

The eventual cost of a 100-passenger "Mars-liner" is argued to be:

a. $5 Billion for a 25 MWe solar electric propulsion system at $200/W (100 for array and 100 for propulsion equipment);

b. $3 Billion for a crew module mass for 100 passengers, 600 t. @ $5000/kg.

c. Vehicle value is $8 billion; writeoff per mission $1.6 billion with the writeoff occurring over 5 mission opportunities (12 years). Including cost of money at 8%, the investment is amortized in about 20 years.

d. Resupply is estimated as 800 t. at $1500/kg delivery cost; the resupply cost is $1.2 billion

e. The cost per passenger comes to $2.8 billion/100 = $28 million.

It is presumed that transport service between Mars orbit and Mars surface will be via reusable MEV based on Mars, supplied with propellant produced on Mars. An earlier STCAEM report included a concept study for such a vehicle.

The solar-electric vehicle cost is dependent on array costs which have here been postulated at a value one order of magnitude less than present-day solar array production costs. This reduction can be justified on the basis of production rate and automation.

A rough cost estimate was also made for the nuclear thermal rocket/L2 basing/lunar hydrogen architecture described above. NTP/L2/lunar hydrogen comes out less, at about $20 million, but the lunar hydrogen production cost estimate is highly uncertain, based on a cursory production system concept analysis.

Many people in the world today have earned income (i.e. not investment) with net present value exceeding this amount. These people are apparently "worth" $30 million, if one assumes "efficient markets". The question is, can people be worth that much on Mars?

A major related issue is "what degree of self-sufficiency is reasonably achievable?" The cost to deliver habitation and other facilities from Earth for large numbers of people is prohibitive. Self-sufficiency for a settlement must mean more than life support and propellants; it is a complete functioning economy. This economy is largely undefined: We don't know what it would produce, or how; we don't know what it would import or export; and we don't understand how the economy would work and grow as a function of size or time. To understand the economic benefits and affordability of a settlement program, these questions need to be addressed. While we cannot obtain complete answers with today's knowledge, much can be learned that is presently unknown. One key result of such analysis will be definition of questions that must be answered by research and by Mars exploration itself.

2.6.4 Proposed Approach to Future Program and Architecture Definition

While the appropriateness of human settlement as a rationale for Mars is explored and developed, at the same time it is important to develop plans for the definable future parts of the program, both human and robotic. Mars exploration began a generation ago with Mariner 4 and continues with Mars Observer. Current mission concepts such as MESUR fit into the obvious need to understand Mars better. Reasonable estimates of exploration activities on Mars should be achievable through at least the first few human missions. Figure 2-23 presents a few representative questions, some of which can be answered by study and some by spacecraft or humans at Mars. These are by no means exhaustive or necessarily top priority; they are meant to indicate the nature of questions that lead to a program definition and to suggest developing a hierarchical ordering of questions that leads directly into a program/mission requirements format.

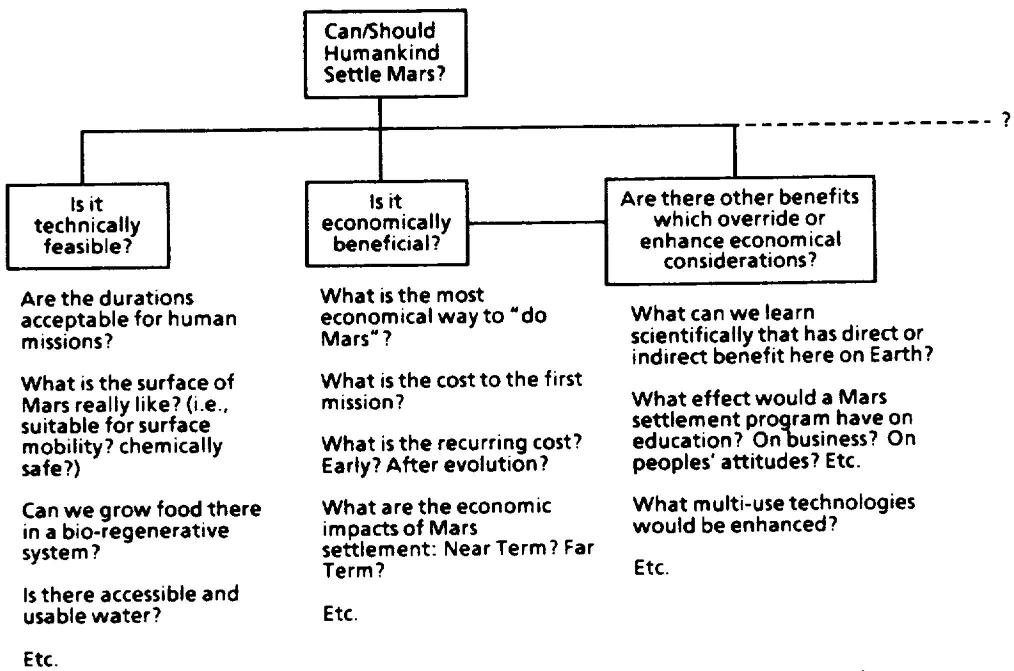

Figure 2-23. Sample Questions and Demonstrations

Mars exploration studies, ever since the von Braun proposal of 1952, have been strong on how to get to Mars and back and weak on what people will actually do on Mars. (Automated mission studies have done a much better job of what their robots will do.) With a defined program rationale and goals, a hierarchy of questions should lead directly to definition of human activities. This, in turn, should enhance public interest in Mars exploration since people easily relate to what people do, especially if it is exciting. Thus rationale development and planning details of actual missions for Mars explorers go hand in hand. As this report and the earlier STCAEM reports show, there are many acceptable ways for humans to get to Mars and back. It's urgent to concentrate on what happens when the human explorers arrive on the red planet.

A settlement program has open-ended goals but can have strictly bounded commitments; that is, it can be defined and approved piecemeal, with each approval commitment having defined goals, costs, schedules, results, and future options. We already know that early steps towards self-sufficiency will reduce the cost of human operations on Mars A settlement program could have an end point as limited as a few conjunction-class missions with food and propellant production. If things go well, a settlement program, after many incrementally-approved segments, could be as ambitious as millions of people on a terraformed Mars.

In either case, the first few decades of the program, the part we can define now, are similar.

The definable phase is intended to establish the feasibility and economics of settlement. It consists of answering questions and conducting demonstrations. The questions can be placed in a hierarchical order conducive to producing program requirements. A program plan produced by this approach has several virtues:

a. The questions and demonstrations can be economically ordered based on value of knowledge provided and cost of obtaining it.
b. The issue of human versus robotics missions and the program schedule fall out from this.
c. It is neither necessary nor desirable to set arbitrary dates for missions.
d. The technology program can similarly be economically derived.

2.7 Architecture Assessment Concluding Thoughts

STCAEM created a comprehensive set of transportation design and operations concepts and a supporting data base for all foreseeable phases of a human exploration program, using technical tools and techniques which did not exist when similar studies were done in the 1960s. The study emphasized rationales and conclusions, in order to understand the meaning of results. The STCAEM study is complete, but the exploration of Mars has just begun. This section has concluded with a discussion of what was learned and where to go from here:

a. Without an adequate program rationale, there is no Mars program fast enough, cheap enough, or "better" enough to be funded and stay funded.
b. STCAEM probably did not find the best possible architectures.
c. What is "best" is not knowable without rationale.
d. It is essential to have a rationale and program plan tightly linked to national goals and strategies.
e. Producing these is more difficult work than designing architectures but far more important!

3.0 ABORT ASSESSMENT FOR MARS TRANSPORTATION ARCHITECTURES

Interplanetary mission design entails complex analyses and trade studies involving trajectory design, Earth ground operations, life support, and other vital concerns. One important part of this mission design process is mission abort planning. Abort planning involves choosing an abort strategy that enables attainment of scientific and exploration objectives while enhancing crew safety to the maximum extent practicable. As a part of 1993 study activities, STCAEM conducted an abort assessment investigating conjunction class missions that utilize "to-surface" abort strategy. It was found convenient to compare abort-to-surface strategies with more conventional strategies that include missions with flyby aborts and on-orbit abort. Section 3.1 defines the abort strategies that were analyzed in this study.

This report summarizes the results of the analysis which accomplished a semi-quantitative assessment of mission design impact on Mars architectures that utilize abort-to-surface strategies. Emphasis in this study was placed on conjunction missions with abort-to-surface mission strategy[1]. In particular, this study looked at the 2009 ExPO reference mission[2] with comparison data provided for the follow-on 2011 mission and the 2018 mission. Note that the 2011 mission opportunity is a "hard" year in the Earth-Mars 15 year synodic cycle and the 2018 year is the "easy" year in the cycle. "Hard" and "easy" refer to the round trip energy requirements.

3.1 ABORT APPROACH FOR MISSION ARCHITECTURES

An earlier phase of STCAEM identified Mars flyby aborts for 2014-2020 opportunities[3]. The abort trajectories identified in this previous study were for NTP mis s that assumed "all-up" manned phases: included in the outbound phase on the manned mission are the capture and return propellant, CRV, an outbound and return habitat, and extra consumables for abort events that require extended in-space stays. Abort issues are more involved for "split" manned transfer (e.g. the ExPO reference) than for "all-up" missions. The greater complexity is due in part from the distributing the pieces of the total mission among several Mars flights.

3.1.1 Abort Definitions (to-surface, to-orbit, flyby, events, etc.)

Several abort strategies are referred to in this report. These are "Abort-to-Surface", "Abort-to-Orbit", Flyby, and "Return-Next-Chance ". Conjunction class missions are generally planned with Abort-to-Surface and Abort-to-Orbit modes. Return-Next-Chance with Abort-to-Orbit is the typical abort planning scheme for an opposition class mission.

A strict Abort-to-Surface is a mode of abort that was adopted by the ExPO for their reference mission. Abort-to-Surface devotes as few resources of the mission as possible to up-front contingencies for abort. If some event occurs that normally requires abort, the strict Abort-to-Surface mission philosophy dictates that the crew go to Mars surface and if possible return to Earth at the end of the mission, or continue on the surface until a rescue can be mounted. This rescue could use the next nominal mission re-configured to perform the necessary rescue function. If a catastrophic event occurs that precludes the crew from remaining on surface to end-of-mission (EOM), the mission and crew will be lost.

The Abort-to-Orbit mission mode is usually in combination with the other abort modes. Abort-to-Orbit simply allows for sufficient consumables on-board the MTV or the ETV to sustain the crew in case of certain abort events. Thus, if some event dictates that the crew can not go to the surface after a nominal arrival, they can remain on-orbit until an EOM return. Another example that would call for an Abort-to-Orbit could be an early return to orbit, requiring an an EOM return or a rescue mission.

All-up missions generally employ the Return-Next-Chance abort strategy. With all-up mission contingencies, the crew can stay in orbit, on the surface, or in some cases immediately return, depending on the the time and characteristics of the abort event. The Return-Next-Chance strategy can increase the the number of possible ways to recoup from an abort event, and therefore the Return-Next-Chance strategy may be more flexible.

3.1.2 Mars Architectures and Standard Abort Modes

Several Mars transportation architectures are examined in this abort analysis: Nominal NTP, Cryo-Aerobrake (Cryo-AB), Mars Direct, ExPO Reference, Modified ExPO, and a nominal electric. The standard abort modes for these architectures are described in figure 3-1 and the discussion below.

Architecture	Mission Type	Abort Approach
Generic NTP (e.g. Synthesis)	Opposition	Return to Earth whenever possible
Cryo AB (e.g. 90 day)	Opposition	Return to Earth whenever possible
Mars Direct (ala Zubrin)	Conjunction	Abort-to-Surface
ExPO Baseline	Conjunction	Abort-to-Surface (strictly)
Modified ExPO	Conjunction	Abort-to-Surface/Abort-to-Orbit
Generic Electric (STCAEM)	Conjunction or Opposition	Abort-to-Surface or return on reduced power

Figure 3-1. Architectures and Abort Options

3.1.2.1 Nominal NTP and Cryo-AB. The first nominal NTP mission, as delineated in the Synthesis Report[4], and the cryo-A/B mission, as defined in the 90 day study[5], are opposition class missions with Mars stay times of 30 to 100 days. Opposition missions are possible every 2.2 years and therefore, rescue missions are can be mounted on roughly two year intervals.

The conventional abort strategy chosen for these two architectures is Return-Next-Chance. If an abort event that is not related to main vehicle propulsion occurs before the nominal departure (within 30 to 100 days), the opposition class vehicle has adequate fuel to depart early. If the abort event precludes an early departure, the crew can go to the surface or remain in orbit until the a rescue mission can be mounted. Another option that may be possible if the crew can not leave on a nominal departure would be to modify the parking orbit period to leave at the following opportunity.

3.1.2.2 Mars Direct, ExPO Baseline, Modified ExPO. Mars Direct, ExPO Baseline, and the Modified ExPO mission are conjunction style missions. Thus, the missions are approximately 2 years apart with long stay times (500-600 days) between return opportunities. Each of these mission architectures employ abort-to-surface in their mission design. Mars Direct and the ExPO Baseline have an abort mode of strictly Abort-to-Surface, but the Modified ExPO additionally employs Abort-to-Orbit and Mars flyby.

One of the disadvantages of the conjunction style mission is related to rescue opportunities. The return opportunity falls several months before the next mission arrival from Earth. This return constraint is related to the physics of interplanetary transfer. Thus, conjunction arrival/return opportunity constraints aggravate the abort scenario by requiring additional living space and consumables for the rescue crew over the duration of another opportunity (approximately 2 years, including transfer time). Consumables concerns are further addressed in section 3.6.

3.1.2.3 Generic Electric (Propulsion). The generic electric mission can fall into the category of conjunction or opposition type missions because of the flexible characteristics of electric propulsion. An electric mission can be either nuclear powered or solar powered.

The abort approach employed in nominal electric mission is Abort-to-Surface or return on reduced power. A reduced power return will entail a longer return trip, but, the windows of opportunity are significantly wider than a conventionally powered mission. If an Abort-to-Surface is required, then the electric mission will incur the same consequences as described in sections 3.1.2.1 and 3.1.2.2.

3.2 ABORT FLOW FOR TYPICAL MISSIONS

In this section, abort flow for the generic NTP, ExPO baseline, and the generic electric are compared. This comparison will provide the general characteristics of the each flow and also indicate the similarities and differences between the abort modes of the architectures.

3.2.1 Generic NTP

The generic NTP mission, as indicated in figure 3-2, is broken into nine primary events: Trans-Mars Injection (TMI), Early Trans-Mars Coast, Late Trans-Mars Coast, Mars Orbit Capture (MOC), Prepare for Descent, Descent, Surface Mission, Ascent, Trans-Earth Injection (TEI), and Earth Aero Entry. This delineation of the mission events was chosen for clarity and convenience to illustrate certain abort modes.

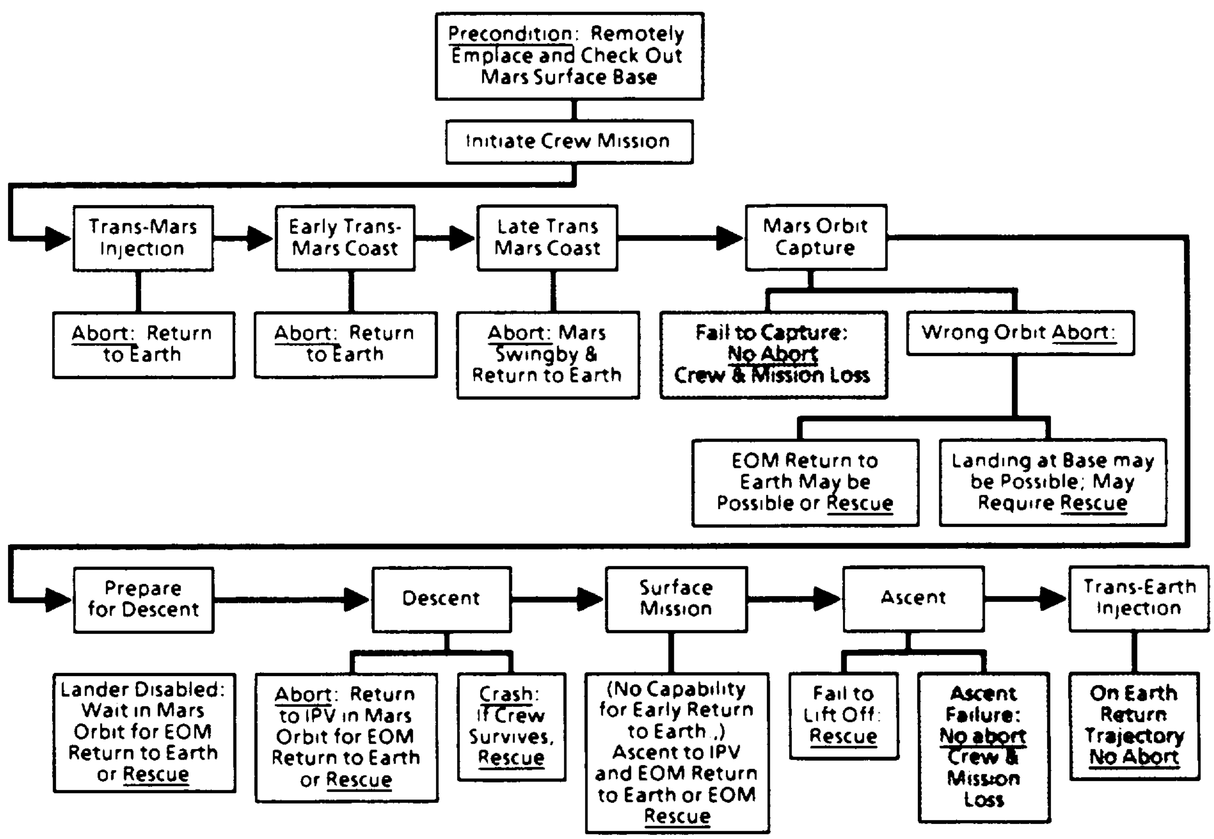

Figure 3-2. Generic NTP Abort Flow

Under each of the primary events of shown in figure 3-1 are one or two typical abort events. For example, under the Early Trans-Mars Coast is an event that indicates that an anomalous event has occurred, such as transfer habitat malfunction, precluding long-term use. The abort mode for this event dictates immediate return to Earth. Note that this event cannot be complete loss of propulsion because there can be no immediate return without propulsion. Each event under a primary event is either shaded gray or is not shaded. A non shaded event indicates that there exists a way to abort the event, and the shaded events indicate that no way to abort has been identified or enabled. For all subsequent abort flow charts, this shading convention is used.

This discussion does not purport to exhaust possible abort related events. The anomalous events designated as "No Abort" cases were assumed not to have an abort (a) because of the prohibitively high cost in delta-V required to correct the trajectory, target an Earth return trajectory, or (b) the abort event has no known way of escape.

For this generic NTP mission, there are three primary events that have no abort scenarios. First, if the MOC maneuver fails to occur correctly, the vehicle could fail to capture and the mission and crew would be lost. Second, if ascent failure occurs after lift off the crew could crash or miss rendezvous with the return vessel, again resulting in mission and crew loss. Third, if the TEI fails in such a way that the vehicle is placed on an interplanetary trajectory that does not intercept Earth, or if a habitation failure occurs during the return transfer, the mission and crew will be lost.

3.2.2 ExPO Reference

For the ExPO Reference mission flow, figure 3-3, the primary events are identical to the corresponding events of the Nominal NTP mission, but the secondary abort events are different. Note the increase in the number of anomalous events in which no abort has been enabled. It should be noted that this is true for the same abort events that were identified in the generic NTP mission.

An explanation of this reduction in number of events that have aborts is found in the abort strategy for the reference mission: Abort-to-Surface. As was previously described, the reference mission places almost no abort contingencies in the manned phase of the mission, other than on the surface. Thus for the first three primary events, TEI, Early Trans-Mars Coast, and Late Trans-Mars Coast, there is no CRV on the outbound vehicle, precluding an Earth return. For MOC and Prepare for Descent, there are not adequate consumables on board the outbound vehicle for a stay in orbit until a rescue could be mounted on the next opportunity. The surface phase of the mission assumes abort-to-surface. This abort philosophy is inherently an effective abort approach once the crew is on the surface, but it should be pointed out that there are not adequate consumable on the Earth return vehicle in the event the crew wishes to go to orbit early to await rescue or EOM return.

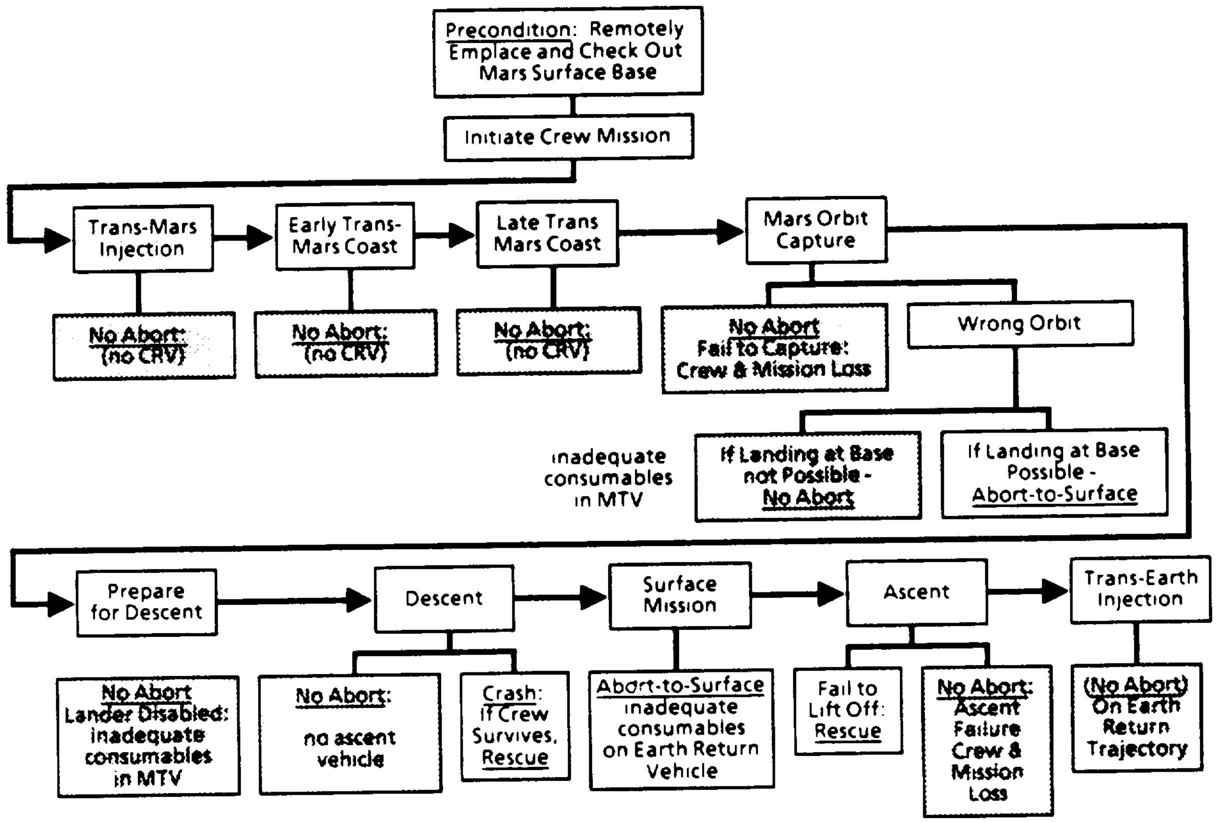

Figure 3-3. ExPO Baseline Abort Flow

3.2.3 Propulsive versus Aerobrake Mars Capture

The ExPO reference has two options for capture at Mars. The first option is propulsive capture with, most likely, an NTP system. A propulsive choice of capture can make sense for the mission because the NTP is used for TMI and can therefore easily (and perhaps economically) be used at Mars capture. An additional reason that a propulsive Mars capture would be effective for the manned phase of this mission is that there will be abort propellant on board for a powered Mars flyby or a non-powered flyby with a deep space maneuver on the return leg. The second option is aerobrake capture at Mars. Aerobrake capture can reduce IMLEO by eliminating the fuel needed at Mars for capture. Also, aerobrake descent is required for Mars landing, allowing for the savings that a common capture and descent brake may bring.

In figure 3-4, propulsive and aerobrake Mars capture are juxtaposed to contrast differences in their abort capability. There is no difference in the number of potential aborts unless the CRV is included on the outbound vehicle. For the aerobrake mission, if the CRV were included on the outbound vehicle, the first three primary events generally have no abort because of no propulsion system on the aerobrake case. An exception to this is a free return Mars flyby abort for Late Trans-Mars Coast (if a free return is possible). Note that "common" in figure 3-4 means that the "No Abort" is true for the propulsive capture and the aerobrake capture. Also, given the strict split strategy of the ExPO baseline mission, there is no ascent vehicle on the manned outbound mission to Mars, resulting in a "No Abort" for the Descent phase of the mission It is pointed out also that if something goes amiss in the MOC and the base can not be reached with the descent vehicle, then no abort is possible since the crew does not have adequate consumables on board to maintain them until a rescue or EOM return can be executed.

Abort Event Phase	Mars Propulsive Capture	Mars Aerobrake Capture
Trans-Mars Injection	No Abort (no CRV)	No Abort (No CRV or propellant)
Early Trans-Mars Coast	No Abort (no CRV)	No Abort (No CRV or Propellant)
Late Trans-Mars Coast	No Abort (No CRV) Free Return (if available), Powered Flyby, Non-powered flyby/DSM	No Abort (no CRV or propellant) Free Return (if available)
Mars Capture (fail to capture)	No Abort (common)	No Abort (common)
Mars Capture (Wrong orbit)	No abort if surface base unreachable (to -surface only)	No abort if surface base unreachable (to -surface only)
Prepare for Descent	No Abort if descent vehicle disabled (no provisions on orbit)	No Abort if descent vehicle disabled (no provisions on orbit)
Descent	No abort if an ascent vehicle is not part of lander (Split)	No abort if an ascent vehicle is not part of lander (Split)
Surface Mission (need to leave surface early)	No abort (to-surface only) (no provisions on return vehicle)	No abort (to-surface only) (no provisions on return vehicle)
Ascent Failure	No Abort (common)	No Abort (common)
Trans-Earth Injection (miss target return velocity)	No Abort (common)	No Abort (common)

Figure 3-4. Propulsive verses Aerobrake Capture at Mars

3.2.4 Generic Electric

An abort flow chart for the generic electric mission is found in figure 3-5. This chart indicates that there are fewer primary events for an electric mission than a conventional mission. For example, the outbound phase and the capture phase are combined into one event block, the first primary event shown in figure 3-5. Therefore there are fewer events that can go awry. However, the likelihood of a partial loss of power, leading to a return to Earth on reduced power, is estimated to be high compared to the likelihood of a failure in a high-thrust propulsion system.

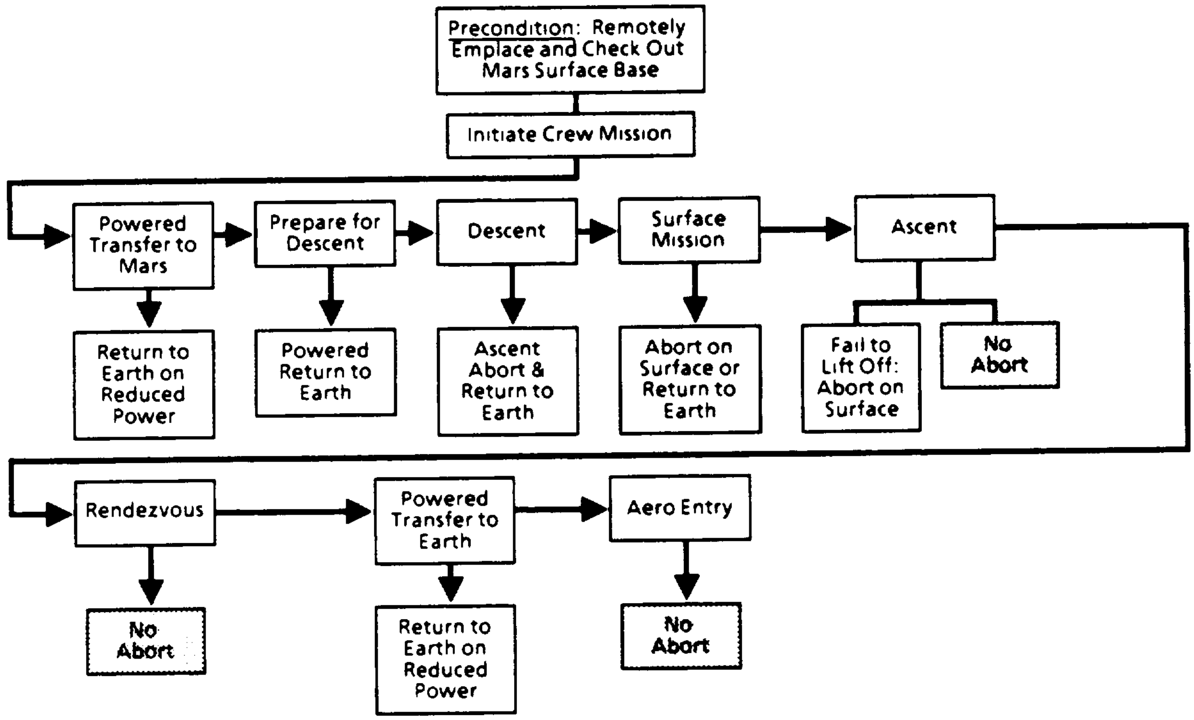

Figure 3-5. Generic Electric Abort Flow

3.3 RECOMMENDED ABORT FIXES TO THE ExPO REFERENCE MISSION

3.3.1 Abort Fixes Summary

An abort in this study is classified as relating to trajectories, hardware, and consumables. We make recommendations in this section that can be thought of as adding abort options in the abort strategy for the NASA ExPO baseline mission. These recommendation are made with the goal in mind of keeping the Abort-to-Surface philosophy intact, but planning for contingencies outside of the strict Abort-to-Surface mission mode. These contingencies are not considered too "expensive". Most of the expensive abort scenarios involve providing high delta-V trajectory response to the abort event. Our recommendations avoid planning for high delta-V (high IMLEO) trajectories in abort scenarios.

The "fixes" listed in figure 3-6 directly affect the mission in terms of the above mentioned abort classification. For example, the first of the abort fixes is to place a CRV on the outbound manned phase of the ExPO mission. If the CRV is not needed, it may be jettisoned before descent. Note that the CRV could serve as a supply cache in the event of an on-orbit abort. This CRV inclusion in the manned phase is a hardware fix to the abort deficiency. Another possible abort fix for this primary event could involve a rendezvous of the manned Mars transfer vehicle with the previously placed return vehicle. A rendezvous is a trajectory answer that may not be too expensive.

Additions to Abort Strategy	Correction Type
Place a CRV on piloted Mars transfer vehicle or allow for co-orbit capture and rendezvous with the Earth return vehicle.	Hardware Trajectories
On-orbit abort mode included in abort strategy • "stay-time" consumables should be on Mars transfer vehicle* • "stay-time" consumables should be on Earth return Vehicle* • consumables adequate for a free return or powered flyby	Consumables
Abort from descent contingency included in abort strategy • requires integrated descent/ascent vehicle on manned mission • allows abort to ETV	Hardware
Provide propulsive capture for manned Mars transfer vehicle. • may provide early trans-Mars abort • provides Mars flyby abort, powered or DSM	Hardware trajectories
*3-burn Mars departure if rescue vehicle rendezvous with Mars transfer vehicle.	

Figure 3-6. Additions to Abort Strategy

3.3.2 Qualitative Impact of Fixes (Modified Baseline Abort Flow)

By incorporating the fixes into the abort strategy, a very significant increase can be made in the number of enabled aborts. It was shown in section 3.2.2 that there were 8 out of 13 abort events that the ExPO mission did not allow some method of recovery. The modified ExPO baseline mission flow shown in figure 3-7 indicates that the propulsive capture case reduces the no recovery incidences to 3 out of 12. On the same figure, the aerobrake capture case is also shown. For aerocapture, there remains no abort for the TEI and Early Trans-Mars coasts because of no propulsion. But the remainder of the aerobrake mission has the same reduction in non-recoverable aborts as the propulsive capture case. One other point that should be noted from figure 3-7 is the reduced chance of mission and crew loss due to a propulsion failure at TEI. Additional consumables on the ETV means that in the event of TEI failure, the crew can survive until repairs are made or a rescue can be mounted for the next opportunity.

3.4 MISSION FAILURE CATEGORIES

An event leading to an abort can result in several outcomes. First, corrections could be made with no mission or crew loss. For example, the crew could abort-to-surface, the mission could be completed, and a successful rescue is undertaken. The second case to consider consists of a loss of mission, but the crew is returned safely to Earth. An example of this kind of abort could entail the following scenario: Habitat on Mars is remotely detected as having irreversibly malfunctioned, the crew conducts a Mars flyby, and the subsequent return to Earth is successfully completed. This abort scenario entail a mission loss but the crew is safely returned to Earth. A third case that is considered could be a crew loss (e.g. ascent vehicle misses rendezvous with Earth return vehicle). Of course, if the crew is lost, then the mission is defined as lost even if all mission experiments are completed and all data are returned to Earth.

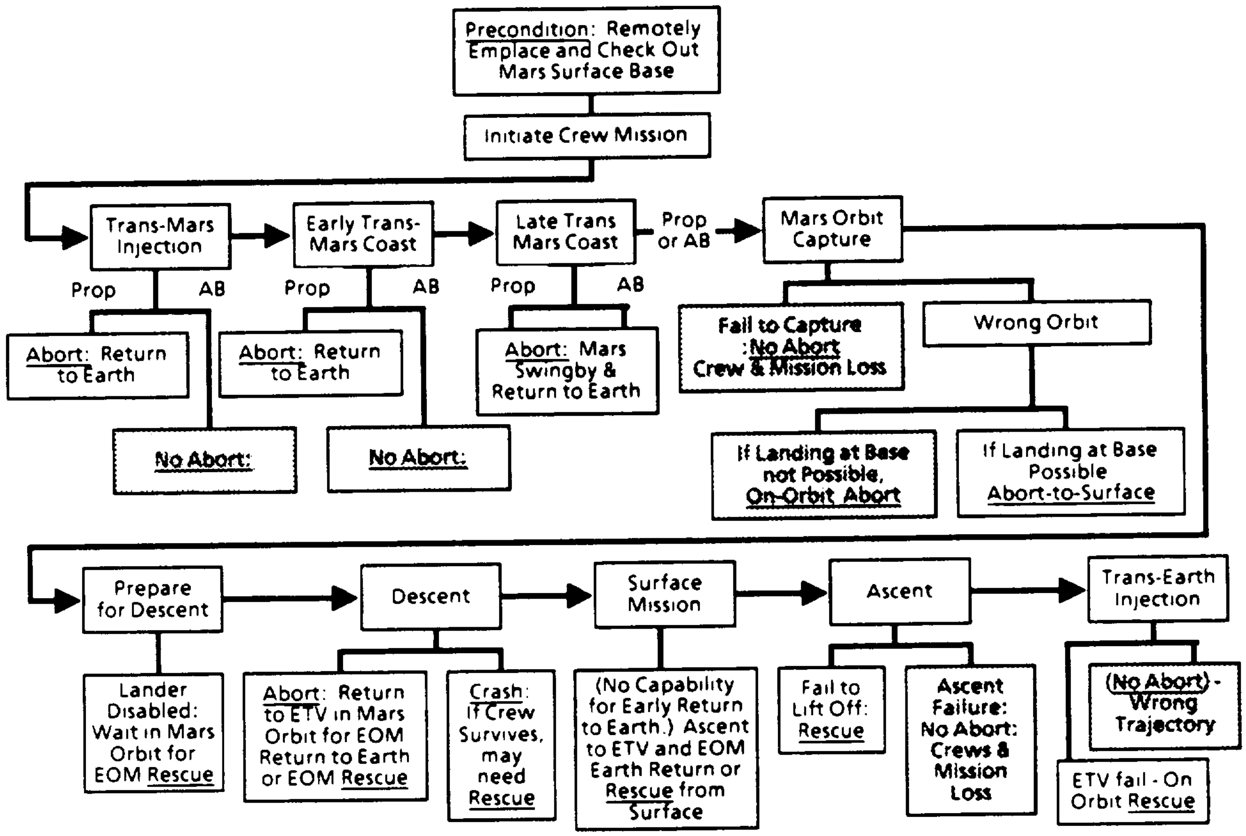

Figure 3-7. Modified ExPO Baseline Abort Flow

Therefore, the data shown in this section is differentiated into the three categories, viz. "mission loss", "crew loss", and "mission success". These three categories are taken as mutually exclusive events, simplifying the probability computations.

In this section, we show results of an investigation into the relative probability of mission failure for four mission modes that have been studied in this report: generic NTP, ExPO Reference, Modified ExPO, and generic electric, see figure 3-8. Relative probability of failure implies that the validity of the data used to create figure 3-8 only has basis in a *comparison* of the missions. It is not intended for use in measuring the absolute chances of success of any particular mission.

The salient results of this mission failure analysis can be summarized in the following points:

a. The generic NTP mission has the lowest overall probability of mission failure.

b. The ExPO has the highest probability of crew loss and next to the highest overall probability of failure.

c. The modified ExPO mission has slightly greater overall probability of mission failure than the generic NTP and significantly less chance of crew loss than the ExPO reference.

d. The generic electric mission has the lowest probability of crew loss and the highest overall probability of mission failure.

Thus, the crew and mission safety can be enhanced by making the changes to the ExPO reference abort strategy recommended in section 3.3. Another point of interest in the figure 3-8 is the greater overall chance of failure of the generic electric mission. Electric propulsion technology is in the early stages of technology maturity for interplanetary missions and required operation times are long. The probability of failure for each of the primary propulsion events of the mission is estimated to be higher than the analogous event for conventional missions.

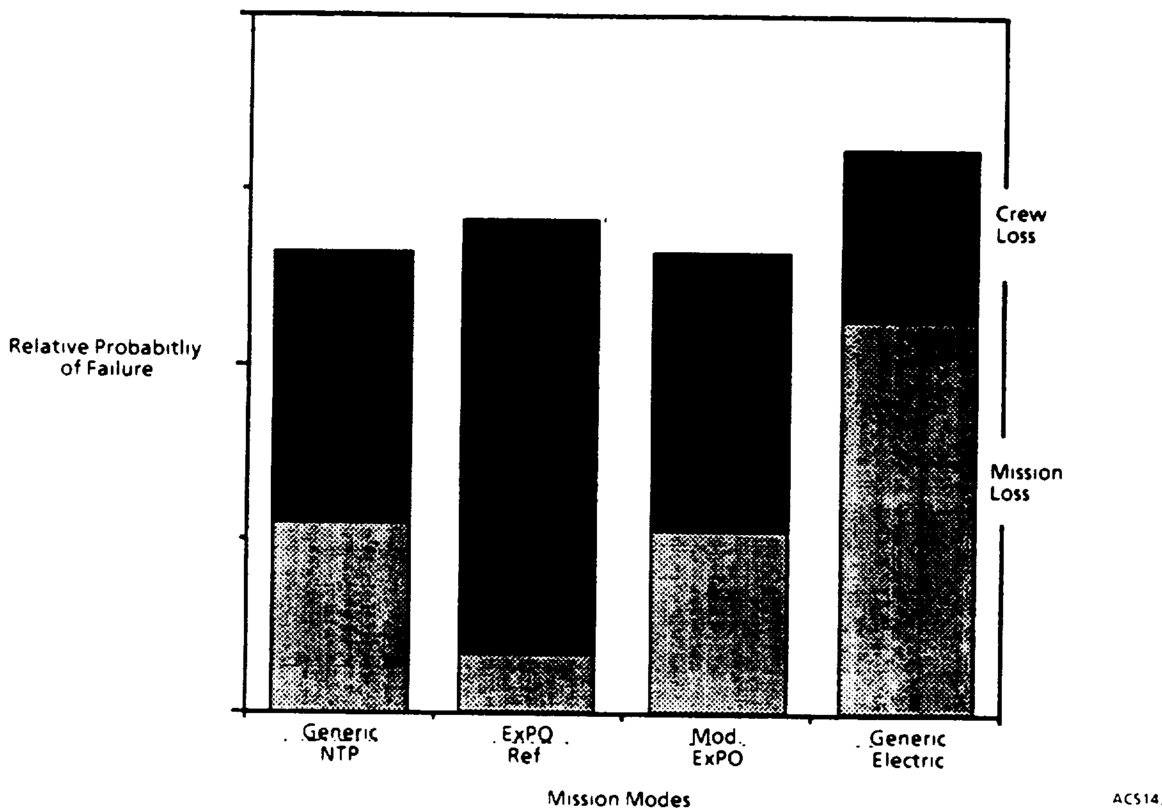

Figure 3-8. Mission Failure Estimates

3.5 TIMING OF MISSION AND ABORT OPPORTUNITIES

Mission opportunities are comprised of cargo missions, the manned mission phases, as well as abort and rescue missions. This section shows on a time line the opportunity phases that we identified. In particular, the missions that are included in this part of the study are the 2009 ExPO reference and a nominal follow-on mission in 2011.

3.5.1 Cargo Missions

The 2009 reference mission time line is shown in figure 3-9. The three cargo missions supporting the 2009 mission are shown with the title "2007 Cargo Missions". These missions leave about 2 years in advance of the manned mission in 2009. These cargo missions traverse type II, low energy, trajectories with a launch window of approximately 40 days. Thus the cargo launch must be launched on a short 13 day centers. Likewise, the cargo mission supporting the 2011 mission are launched 2 years in advance in 2009. The launch window is somewhat wider in 2009 than in 2007, providing approximately 16 day launch centers. Note that the cargo missions supporting an aborted reference mission are shown in figure 3-9 as the third set of cargo missions. Faster cargo missions can be flown in the event of an aborted mission, but the cargo will reach the stranded astronauts well after the nominal departure time. An exception to this late cargo arrival can be made if the outbound leg of the "Reference Abort 1" is used as a cargo mission. Abort 1 would allow the cargo to arrive near the time of nominal departure; at this time the ground supplies may be nearly depleted. It should be noted, however, that Abort 1 must be launched from Earth approximately one third into the nominal mission stay. Therefore, the need for this cargo mission must be known very early in the mission for launch to be achieved. In addition, an Abort 1 cargo mission would be a relatively high energy cargo mission because of its opposition profile (Venus swingby on leg to Mars) on the Earth/Mars leg.

3.5.2 2009 Reference Mission

The characteristics of the 2009 reference mission are presented here along with the related abort and rescue opportunities. The actual launch time, trip times, and stay times are those provided by the ExPO Mars Transportation Working Group. Boeing ascertained the abort opportunities associated with this mission. The 2009 mission departs Earth on October 30 of 2009 and arrives 180 days later on April 28 of 2010. The nominal stay time is 540 days on the surface of Mars. Mars departure is October 20 of 2011, with a 180 day return ending on April 17 of 2012. Total trip time is 900 days.

3.5.2.1 Flyby Aborts for the 2009 Opportunity. Several flyby abort opportunities were identified in this study of the manned phase of the the 2009 mission. Those missions are delineated in figure 3-9 as "Free Return" and "flyby with DSM". Of course, the only part of the mission that abort options are provided for is the manned phase of the mission.

Two kinds of flyby aborts are considered for this study. First, if the mission uses aerobrake capture at Mars, the only abort cases that apply are abort-to-surface and unpowered flyby. The flyby must be a free return because of no propulsion for a powered flyby is provided on a mission utilizing aerobrake capture. For total flyby mission

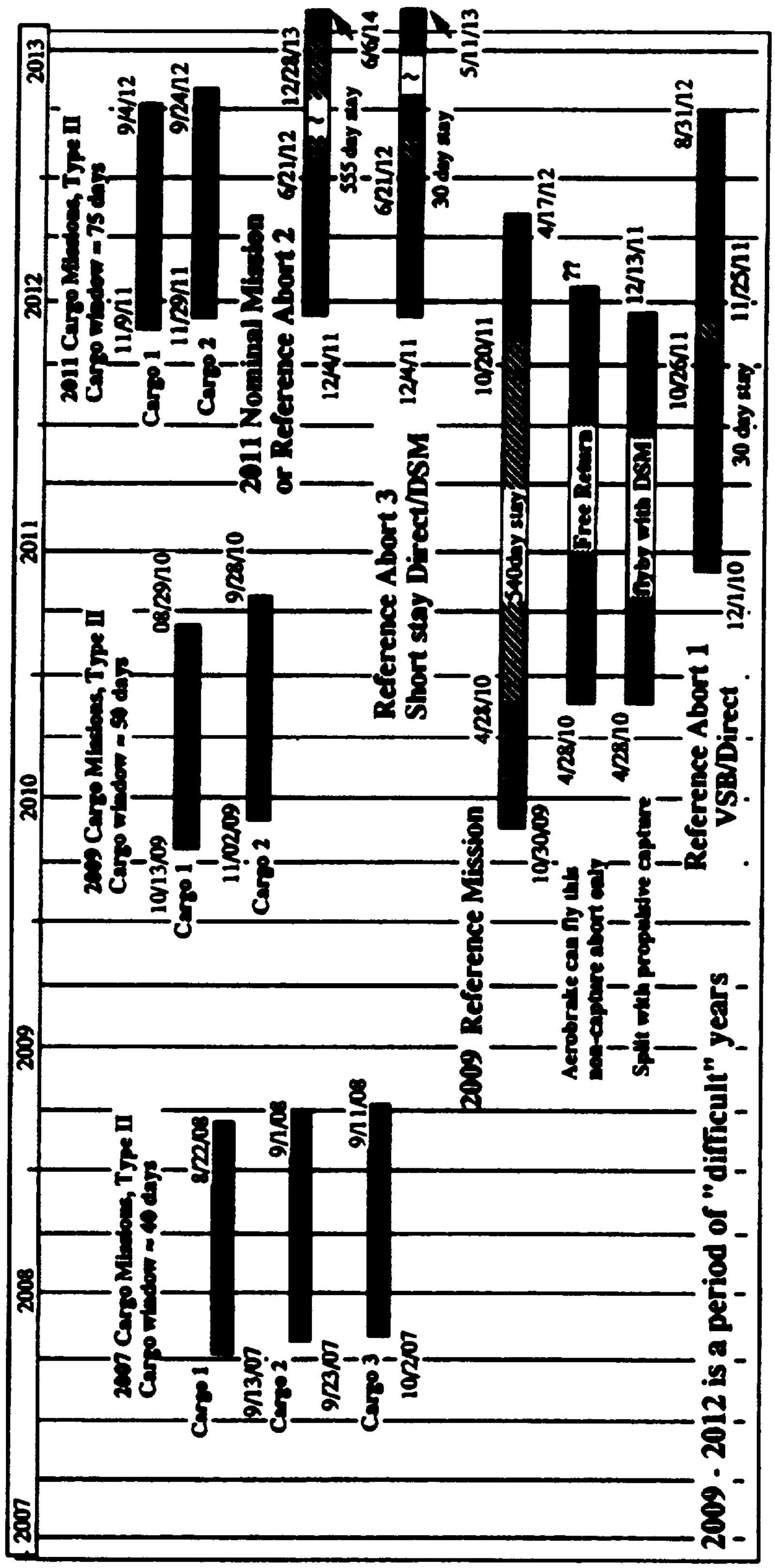

Figure 3-9. 2007/2011 Reference Mission Timeline

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duration of up to nearly three years, a free return flyby was not found for this mission. Yet a free return may be possible[6], hence the question mark at the end of the free return designator in figure 3-9. Second, if the mission incorporates a propulsive Mars capture, then capture propellant and propulsion may be used to perform a powered Mars flyby or, as shown in figure 3-10, a non-powered Mars flyby with a deep space maneuver (DSM) on the Earth return leg may be used. In either flyby case, the long Earth return time (≥600 days) will require significantly more consumables on the MTV than are nominally planned for on the ExPO mission. Flyby aborts also require a CRV for the direct Earth capture and descent. Thus a strict abort to surface requirement precludes the use of flyby abort options.

3.5.2.2 Rescue Missions for the 2009 Opportunity. Rescue missions are delineated in figure 3-10 as "Reference Abort 1", "Reference Abort 2", and "Reference Abort 3". For the remainder of this discussion, the abort missions are referred to as abort 1, abort 2, or abort 3, respectively.

Abort 1 is the earliest rescue opportunity identified for the 2009 mission. It is an opposition class mission, requiring a Venus gravity assist on the outbound leg and a short stay time of approximately 30 days. Note that the return end date is on August 31 of 2012, a 328 day return. This leg of the mission can be shortened from a relatively long type II trajectory to a much shorter type I transfer without a large increase in Mars departure C3. It is unlikely that this abort opportunity will be used, except in very special circumstances, because of the relatively large delta-V required to execute the mission (see section 3.7 for delta-V tables).

Abort 2 is actually the next nominal manned mission in 2011. It is possible to used the next mission as an unmanned or partially manned rescue mission. This mission requires that the crew from the previous 2009 mission and the crew (if any) from abort 2 remain at Mars through another opportunity (until nominal departure in December 2013). This long stay time translates to increases in crew quarters and consumables for those at Mars. Note that abort 2 is a conjunction class mission, requiring less propellant than abort 1.

Abort 3 is an opposition class mission requiring a deep space maneuver on the Earth return leg and a Mars stay time of approximately 30 day. Thus, analogous to Abort 1, Abort 3 is a relatively high energy mission. It would not normally be used for rescue because of the relatively high cost in delta-V (see section 3.7). This mission has the same outbound leg as the nominal 2011 mission (200 days), but utilizes an opposition profile for the return leg and requires 294 days for return.

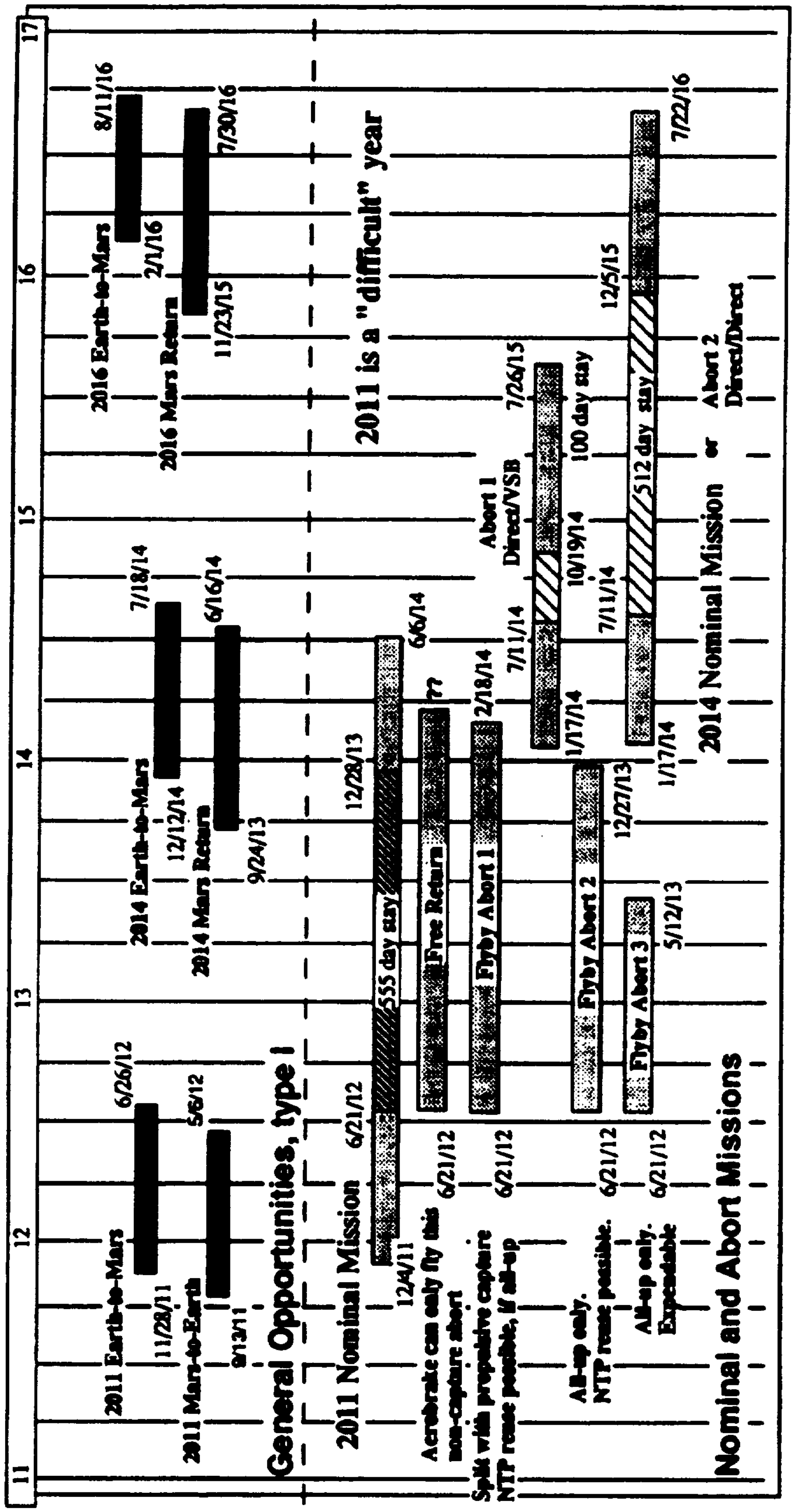

Figure 3-10. 2011/2104 Mars Mission Timeline

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3.5.3 2011 Nominal Mission

The characteristics of the 2011 nominal mission are now described along with the related abort and rescue opportunities. The actual launch time, trip times, and stay times for all of the 2011 cases were were found by STCAEM analyses. The 2011 mission leaves in December 11 of 2011 and arrives 200 days later in June 21 of 2012, see figure 3-10. Note that this is a 200 day transfer (longer than the 2009 outbound transfer) because this mission occurs in the high energy part of the 15 year synodic Earth/Mars cycle. This means that the 2011 transfer time is longer for the same 2009 Earth C3L. The nominal stay time is 555 days on the surface of Mars. Mars departure is December 28 of 2013, with a 160 day return ending on June 6 of 2014. Total trip duration is 915 days. Note that at the top of figure 3-10, nominal type I Mars outbound and Earth return mission are shown for the 2011, 2013, and 2016 opportunities. These type I missions are shown to provide some general opportunity frame of reference to compare with the nominal 2011 mission.

3.5.3.1 Flyby Aborts for 2011 Opportunity. Several Mars flyby abort options were identified in this study for the 2011 mission. These flyby opportunities are delineated in figure 3-10 as "Free Return", "Flyby Abort 1", "Flyby Abort 2", and "Flyby Abort 3".

The two other flyby abort cases, flyby abort 2 and flyby abort 3, can be flown only for missions that have propulsive capture/return propellant and a CRV on the outbound mission. These mission are usually called "all-up" missions because they generally have the MTV and ETV as an integral part of the outbound vehicle. It can be seen from the time line, figure 3-10, that the flyby Abort 2 and Abort 3 are significantly shorter than the free return and flyby Abort 1, but a split aerobraking mission can not perform these aborts (no capture and return propellant on MTV).

3.5.3.2 Rescue Missions for 2011 Opportunity. The 2011 rescue missions are depicted in figure 3-10 as "Abort 1", and "Abort 2". For the remainder of section 3.5.2.2, the abort missions will be referred to as abort 1 and abort 2 respectively.

Abort 1 is an opposition class mission, requiring a Venus gravity assist on the return leg and a short stay time on the order of 30-100 days. The outbound leg of this mission is identical to the nominal conjunction class mission outbound leg. The return end date is July 26, 2015. It is unlikely that this mission opportunity will be used because of the relatively large delta-V required to execute the mission (see section 3.7).

Abort 2 is actually the next nominal manned mission in 2014. As was explained for the 2009 reference mission, it is possible to used the next mission as an unmanned or partially manned rescue mission. This mission requires that the crew from the previous 2011 mission and the crew (if any) from the Abort 2 mission remain at Mars through another opportunity (until nominal departure in December 2015). This long stay time translates to increases in crew quarters volume and consumables for those at Mars. Note that abort 2 is a conjunction class mission, requiring less propellant to perform than Abort 1.

3.6 RESCUE OPTIONS AND CONSUMABLES

3.6.1 Consumables Required for Rescue Operations

The initial October 9, 1992 NASA Mars mission status report from JSC set a consumable allotment for the primary Mars base at 26 tonnes, coming as part of the first three cargo/base buildup flights. In addition a small 6-crew Bio-Chamber was to be another part of the 150 t total initial base mass delivered on these same flights. This was to last for the first manned mission, of approximately 600 days and keep a 2 to 3 year reserve for an abort-to-surface contingency. Using 4 kg per person per day as a general consumable number that includes water, both drinking and hygiene, food, air, packaging and medical and life support spares, a crew of 6 over 600 days uses 14,400 kg (14.4 t) of consumables. The reserves of 11.6 t, if used at the same rate, are good for 483 days; therefore all the consumables cover a total of 1,083 days. For a 2 year abort, the additional stay time beyond the named mission is 730 days (with 10 days additional reserve for possible error), leaving a total of 1330 days that must be covered by surface consumables. There is a short fall of 247 days between what is sent and what is required in case of a failure of the ascent or return vehicles that must be covered by additional resources. Additional consumables could be sent on a direct cargo mission to make up this difference. However, there may or may not be a direct cargo supply flight that will arrive in the 483 days available, depending on when the condition that prevents return is discovered. The timing of the flight opportunities was shown in figure 3-10.

While the NASA study estimated that the bio-chamber could support the crew for the entire 600 day mission, there is some question at to whether this is a reliable resource. It is true that it will be very beneficial to supplement the delivered consumables for psychological reasons, if nothing else; but the first bio-chamber has the possibility of crop failure, a breakdown in biological isolation, or mechanical failure. It appear too great a risk to rely on the initial bio-chamber for significant resources on the first manned mission. The percentage of support the bio-chamber could reasonably supply to extend the consumables is not known. The first mission was considered a trial to "debug" the bio-chamber system, so that on subsequent missions, which call for a larger bio-chamber, the techniques for operating the system and the percentage of support it can give would be well established.

3.6.2 Options for Mounting Crew Rescue Missions

As can be seen from the flight opportunities shown in figure 3-9, two possible time frames for rescue exist: a conjunction mission (late) and an opposition mission (early). For each of these opportunities, three kinds of rescue options present themselves: (a) sending an unmanned rescue vehicle, (b) sending a reduced crew (3 crew) rescue vehicle and (c) using the next mission to exchange the crew. An overview of these options can be found in figure 3-11. For the unmanned rescue vehicle it is assumed that no one on the surface is injured or impaired and that all the crew is on the surface (abort-to-surface). The rescue vehicle must be autonomous, have one each of the return transit habitat, engines and return propellant allotment, CRV, lander with ascent and additional consumables for the extended stay. The question here is the degree of autonomy needed and the access to the appropriate capture orbit. If the late option is chosen an additional cargo flight will be needed to support the Mars crew on the surface.

Rescue Option*	Characteristics	Comments
#1, Unmanned Rescue	Autonomous vehicle; no crew impairment on Mars surface.	Requires no space or consumables for additional crew on surface or in Mars orbit; what degree of autonomy is required?
#2, Three-man Rescue	Rescue vehicle manned with 3 crew; may or may not be Mars surface crew impairment;	For part or all of an additional opportunity, provisions and space needed for 3 extra crew
#3, Crew Exchange Rescue	Rescue vehicle manned with 6 crew; may or may not be Mars surface crew impairment;	For part or all of an additional opportunity, provisions and space needed for 6 extra crew

* Rescue approach is to use the next mission's vehicle(s) in a rescue mode.

Figure 3-11. Options for Rescue

With the three-man rescue vehicle all the Mars crew will also be on the surface. Some of the surface crew could be injured. The rescue vehicle must have the return habitat with engines and return propellant allotment, either two CRVs or a single CRV modified for nine crew, either two landers or a single lander modified for nine and additional consumables for nine crew to remain on Mars and return at the next opportunity. The open items here are the capture orbit, additional consumables, and room on the surface and return vehicle for nine crew. An additional cargo flight may be needed.

The crew exchange mission simply replaces the crew that is stranded with the next crew. The original crew may or may not have someone injured and all are on the surface. The return vehicle is a regular six-crew mission with the full return habitat engines and propellant, one six crew CRV, one or possibly two landers and consumables for the Mars original surface crew plus the additional six crew with provisions for the crew remaining on the surface. If two landers/ascent vehicles are brought, an exchange may proceed as part of a regular mission, provided the ascent vehicle is all that failed on the first mission and the return vehicle is operable and in an accessible orbit. If not, the second crew is stranded on the departure of the first crew. In addition to the concerns of the three-manned rescue scenario there is the problem of 12 people on the surface or in orbit for some time (100 to 512 days) that must be supported. Figure 3-12 shows the consumables per mission phase for each of these three rescue modes in both the early and late opportunities. The accessibility of the consumables, their storage and the durations that the aborts impose must be considered in the mission planning and risk assessment.

Mission Type		Outbound Consumables	Stay time Consumables (Orbit or surface)	Return Consumables	Consumables left on surface	Total Consumables
Nominal 2011 (555 day surface stay crew of 6)		4,800 kg	13,320 kg	3,800 kg	—	21,960 kg
Unmanned Rescue	early*	—	2,400 kg	6,720 kg	—	9,120 kg
	late**	—	12,288 kg	5,520 kg	—	17,808 kg
Rescue with 3 personnel	early	2,100 kg	3,600 kg	10,080 kg	—	15,780 kg
	late	2,100 kg	18,432 kg	8,280 kg	—	28,812 kg
Exchange of Personnel	early	4,200 kg	4,800 kg	6,720 kg	19,920 kg***	35,640 kg
	late	4,200 kg	24,576 kg	5,520 kg	19,920 kg***	54,160 kg
Nominal 2014 (512 day surface stay 6 crew)		4,200 kg	12,288 kg	5,520 kg	—	22,008 kg

Note: 4 kg/day per person is an average value for consumables

* The earliest rescue mission opportunity, valid for problems found in the first part of the mission; consists of 175 days outbound, 100 days at Mars, 280 days return

** The earliest rescue mission opportunity valid for problems found in the last part of the mission; consists of 175 days outbound, 512 days at Mars, 230 days return

*** The mission consists of replacing the stranded crew with the next mission, therefore the next mission supplies must be brought/delivered to sustain the 2016 opportunity

Figure 3-12. Consumables per Nominal & Rescue Mission by Phase

3.6.3 Typical Abort/ Rescue Tree, 2011 Mission

In an overview of abort conditions that can be analyzed for risk assessment (probability of failure) several abort trees were generated. A typical abort tree for the 2011 mission after capture at Mars is given in figure 3-13. The abort options 1, 2 and 3

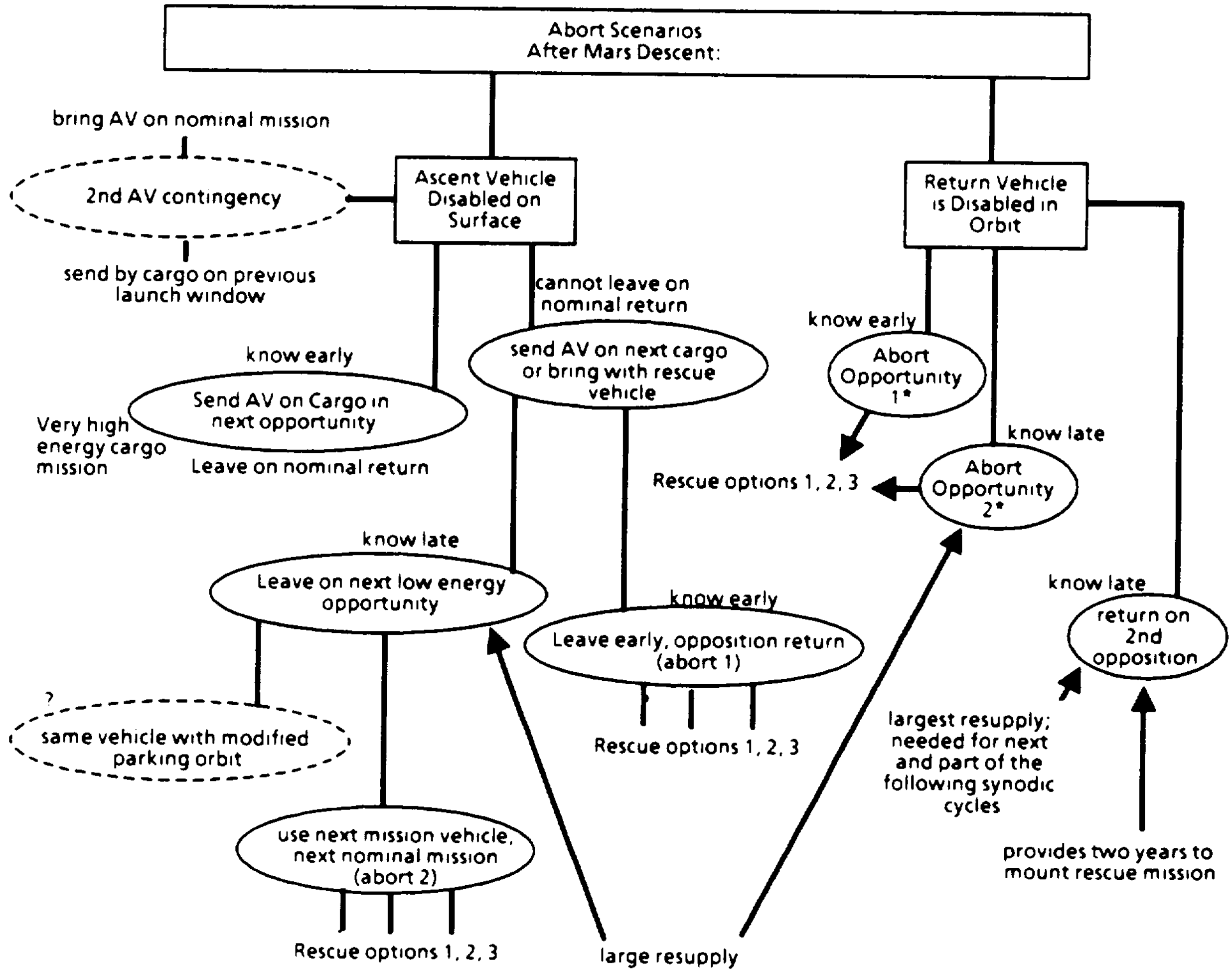

Figure 3-13. Surface Abort Tree

refer to the the options stated above and the opportunities 1 and 2 refer to the opposition and conjunction mission opportunities also mentioned above. These and the flyby aborts can be seen in figure 3-10. The reference to "know early" and "know late" indicates to when a problem is identified in the course of the surface stay and whether there is time to react and use to the next available abort scenario or be forced to choose another path home. This tree covers only the options available if there is a problem discovered on either the ascent vehicle or the on-orbit Earth return vehicle. This can happen well into the surface stay. These discoveries can occur towards the end of the mission surface stay, where when the discovery of the problem takes place can make a difference in available choices. It also points out that it is important to know if the on-orbit Earth return vehicle is operable while the crew is still on the surface. If the fault is discovered on-orbit, then unless there is another descent/ascent vehicle is available, the crew is stuck on orbit and stranded away from the consumables supply. The chart line for the return vehicle disabled shows only the options that exist if the return vehicle condition is known from the surface. Any other condition would have no abort option. The chart lines descending from the ascent vehicle disabled on the surface, indicate that there are more options, if an ascent vehicle can be sent as part of an early cargo flight. Other options and opportunities remain the same as with the return vehicle disabled. Where a large cache or resupply of consumables is needed to perform these aborts it is marked on the chart.

This chart and other abort trees are only the starting point for a thorough abort analysis, which was beyond the scope of the present study.

3.7 ANALYSES METHODOLOGY AND SUPPORTING DATA

Section 3.1 through section 3.6 summarize an analysis evaluating and identifying implications of abort options for several mission profiles. This section describes the technical supporting data for the analyses of the previously mentioned sections.

Vehicle performance data (Earth/Mars launch window and delta-V) is provided for a set of missions that include the 2009 ExPO reference, a follow-on 2011 conjunction missions, and a 2018 conjunction mission. The 2009 and 2011 missions fall in a "hard" part of the 15 1/2 year Earth/Mars synodic cycle. The 2018 mission falls in an "easy" part of the synodic cycle. In this report, "hard" or "easy" means that the mission requires relatively high or relatively low delta-V for a mission of set interplanetary transfer duration. Considered in the study are nominal mission and the type I and type II minimums for the 2011 and 2018 cases. Also investigated in this analysis are rescue mission scenarios that are consistent with an abort-to-surface requirement. Rescue scenarios that are covered included cases that entail direct and Venus swingby trajectories.

Earth and Mars launch window contour data (C3L and Vhp) was generated by Boeing's PLANET code. Further, PLANET generated the trajectory data used in delta-V computations. Listed below is a list of ground rules followed in the analysis of the 2011 and 2018 missions. The 2009 mission profiles were taken directly from data supplied by the Mars Transportation Working Group as listed in the references.

Vhp $\leq$ 7 km/s

C3L $\leq$ 28 km^3/s^2

transfer time = 150 days for 2018 out and return (easy year)

transfer time = 200 days for 2011 outbound (hard year)

transfer time = 160 days for 2011 return (hard year) rescue missions may use combination of direct and VSB trajectories

Early rescue - VSB/Direct , Direct/VSB, VSB/VSB.

Late rescue - Conjunction mission; crew must stay from nominal return to next opportunity

Full-up missions only

The trajectory data and contours were generated for this report by use of two codes: PLANET and MIDAS. These are Lambert-based codes, using "patched conics" to simulate interplanetary trajectories. From the trajectory data for a range of Earth departure, Mars arrival, Mars departure and Earth return dates, Planet was used to generated contours of C3L and Vhp used in the Earth and Mars launch window analysis. It was also used to find optimal elliptical parking orbits and thereby predict the delta-V for the Mars orbit capture and Mars departure maneuvers. PLANET optimizes the parking orbit for minimum departure delta-V by searching over a range of arrival inclinations and periods. This search finds the best arrival conditions for minimizing of apsidal misalignment losses from departure from an elliptical orbit. The misalignment losses include plane change and apsidal rotation required to target the correct Mars departure V-infinity vector. Parking orbit operational constraints such as arrival periapsis lighting angle and arrival periapsis latitude can be included in the parking orbit optimization process.

MIDAS is a NASA code that was used in this analysis to predict the performance required for flyby aborts at Mars. It is a patched conic based program that utilizes a gradient search routine to optimize the trajectory delta-V for a mission. It, however, can not be used as to find optimal parking orbits because it assumes periapsis-to-periapsis burns at arrival and departure. Thus, MIDAS does not take into account apsidal misalignment losses during Mars departure from an elliptical parking orbit.

3.7.1 Launch Window Analysis

Analysis of launch windows consists of determining the time interval that a mission opportunity can be launched. This process includes determining the inclusive dates over which the launches may occur such that those launches meet the mission launch energy constraints. Equivalently, the launch dates are chosen that fall within the launch vehicle payload capability and meet the overall mission requirements. Acceptable launch windows were identified for the 2009 ExPO reference mission, the follow-on 2011 conjunction mission, and the 2018 conjunction mission.

3.7.1.1 2007 Cargo Missions. The 2007 cargo mission supporting the 2009 mission was investigated. This cargo mission was broken into 3 flights with each flown on a type II trajectory. Type II trajectories are defined as transfers that require more than 180 degrees for the mission leg in question. As shown in figure 3-14, the top lobe of the contours represent the type II trajectories and the bottom lobe represents the type I trajectories. The first contour of figure 3-14 shows the launch energy window, C3L, as a dark strip on the upper lobe. From the contour it can be determined that the available launch window will allow 3 launches with approximately 20 day centers. This window, from the second contour of figure 3-14, maintains the Vhp at Mars arrival to a maximum of 5 km/s. A 5 km/s Vhp will allow direct Mars entry with little resulting thermal radiation and minimal concomitant TPS requirements.

3.7.1.2 2009 ExPO Mars Outbound Window. Following the determination of the first cargo mission launch window, figure 3-15 shows the Earth Launch and Mars arrival window for the 2009 ExPO reference mission. The components of this mission are the 2 cargo missions launched on type II trajectories and the manned vehicle launch on a type I trajectory as indicated in the figure. The cargo missions can be launched within a window of approximately 60 days in duration. The reference mission was chosen as 180 day outbound transfer time to Mars. On the figure, the mission optimized for aerobrake is labeled on the 180 day line as AB. Likewise, the all propulsive case is indicated as AP on the 180 day line. The aerobrake case falls farther to the right on the line because the Earth C3L is lower and the Vhp as Mars is higher: optimal for the aerobrake case. This is contrasted with the AP case which falls farther to the right on the 180 day line. The AP case has a higher Earth C3L than the AB case, but has significantly lower Mars Vhp: optimal for the all propulsive case.

The 180 day transfer time yields a window a approximately 35 days for a maximum C3L of 28. If one assumes 200 day returns, as shown by the 200 day line in figure 3-15, the window is only a few days longer, but the Earth C3L drops to a minimum of 20 km^2/s^2 and Mars Vhp can be at a minimum of less that 3 km/s.

3.7.1.3 2009 ExPO Earth Return Window. The Earth return window is shown in figure 3-16. The reference mission has a 180 day, type I Earth return trajectory. A type II trajectory could be a a bit lower in Mars departure C3L, but the transfer time would be greater. Thus, for a type II return trajectory, the in space time that the astronauts must endure would increase. Note that the chosen Mars departure and Earth arrival date corresponds to an Earth arrival Vhp of approximately 8.9 km/s. This Earth arrival Vhp is probably exorbitant in terms of TPS requirements for the return vehicle. A way to reduce this relatively high Vhp can be seen from the figure 3-16. As can be determined

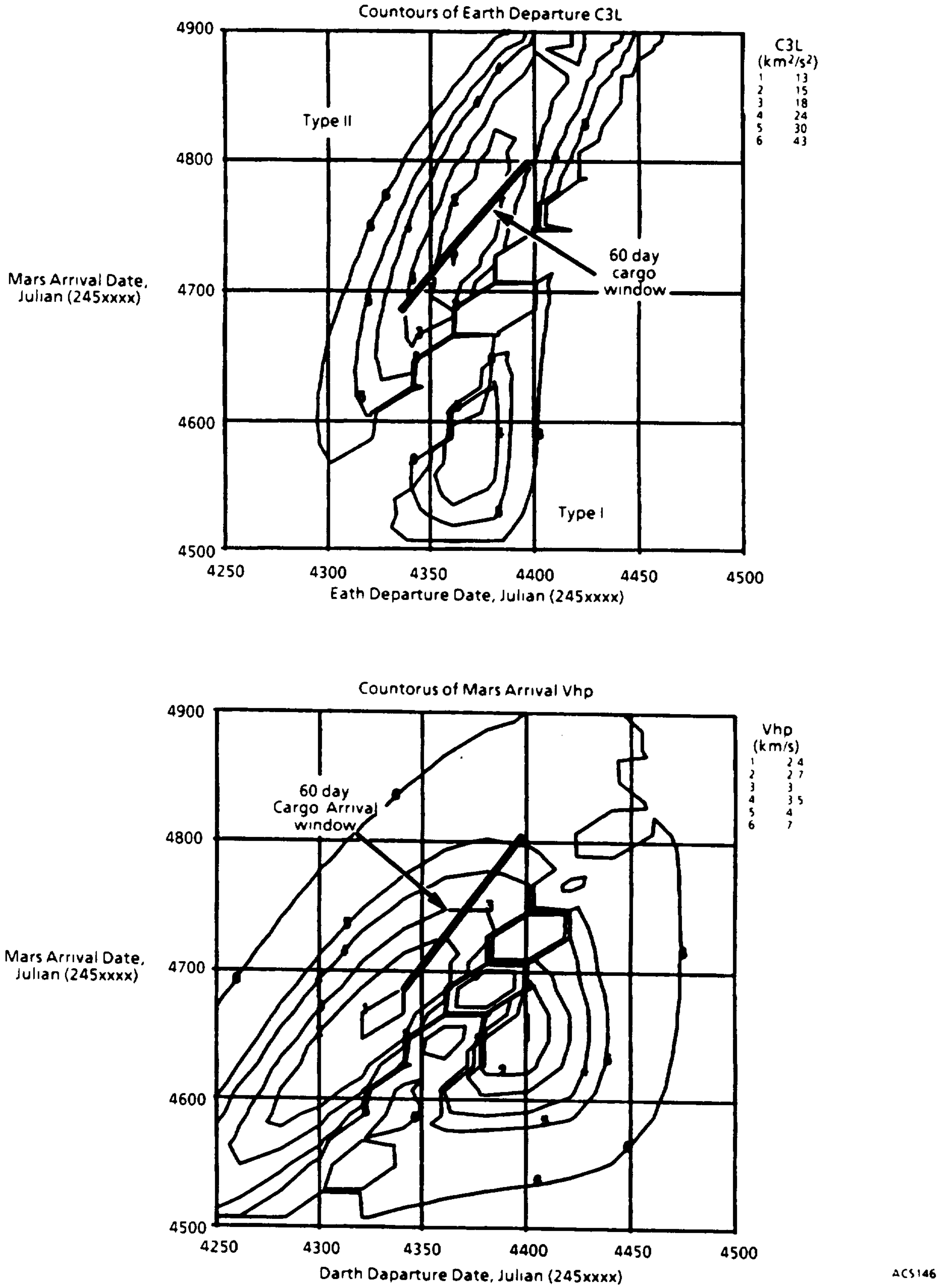

Figure 3-14. 2007 Cargo Windows

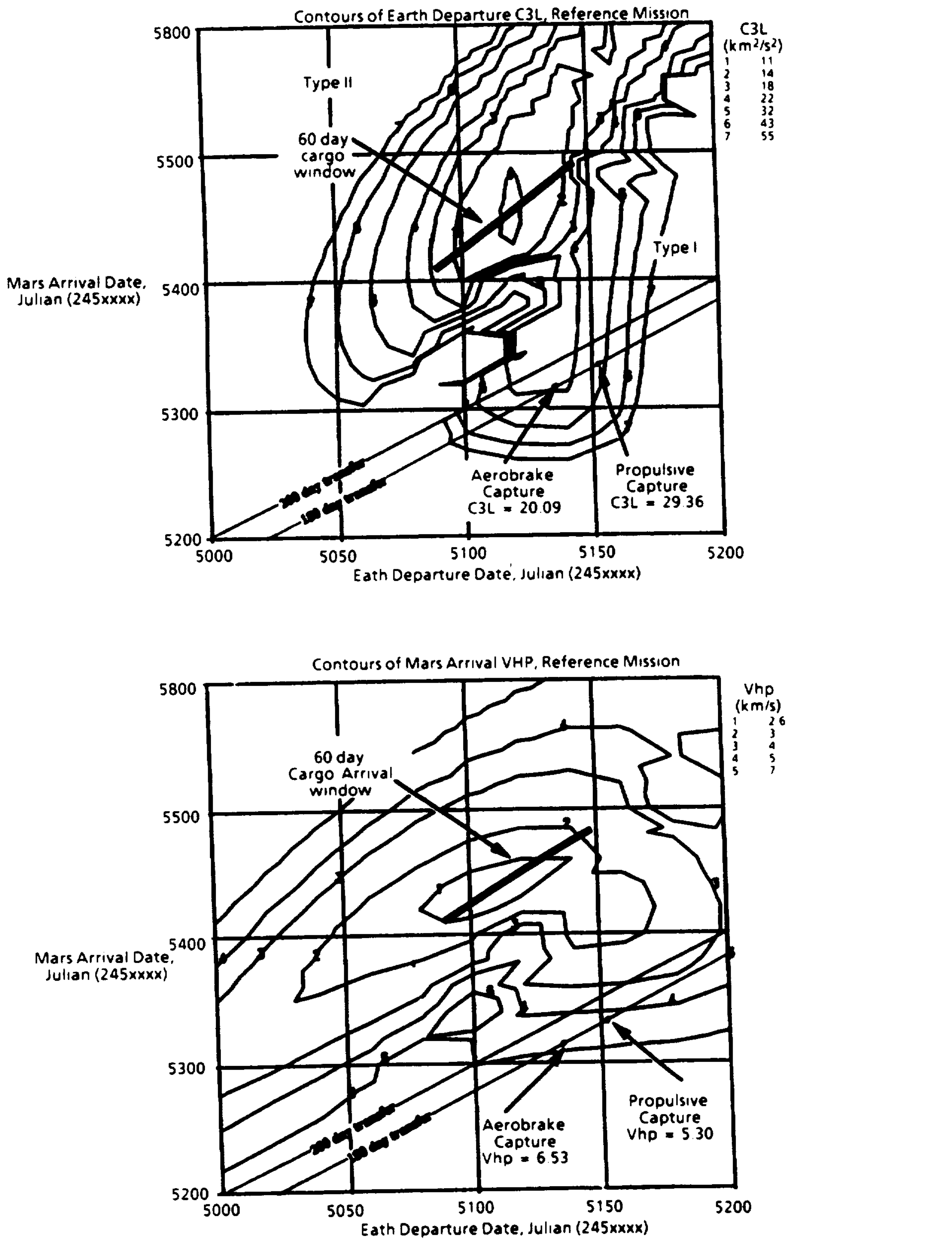

Figure 3-15. 2009 Earth Departure/Mars Arrival Windows

from the Earth return Vhp portion of figure 3-16, this arrival condition could be improved by increasing the Earth return transfer time to 200 days, yielding a return Vhp of less that 7 km/s and requiring only 20 additional days of in-space transfer.

3.7.1.4 2009 Rescue Missions. In this section emphasis is placed on a launch window comparison for two possible modes of rescue for the 2009 ExPO mission. An analysis was performed to show the Earth launch windows and Mars arrival windows for a Venus gravity assist case and a deep space maneuver case. The window analysis was completed for the VSB and DSM cases over an identical and relevant mission space time frame. Note that this window is for a part of mission space that corresponds to the earliest rescue mission feasible from Earth. Practically, this mission could only be used in the case of very early knowledge of the need for a rescue. Given the high delta-V requirements and/or the long transfer times required for these missions, a more likely use of this mission that remains relevant to rescue operations would be to use the outbound leg of this mission to place cargo at Mars in the event that nominal departure can not occur. The cargo can reach Mars before nominal departure of the 2009 mission.

The Earth C3L and Mars arrival contours for the VSB case and the DSM case are shown in figures 3-17 and 3-18 respectively. Some general remarks are in order. The Earth launch windows for the VSB and DSM cases are narrow. For a maximum C3L between 28 and 29, the VSB case has a launch window of approximately 10 days and the DSM case has a window of approximately 15 days. If, however, the requirement for Mars arrival Vhp is 5 km/s or less, the window is effectively reduced to 4 days for the VSB case. For the DSM case, the Vhp limitation effectively precludes missions because there does not exist any part of the window with Mars arrival Vhp of 5 km/s or less.

In particular, figure 3-17 indicates at the dot a possible propulsive capture mission. Note the very narrow window that maintains the C3L below 29 and a low Vhp of just under 5 km/s. If this mission incorporates aerobrake assisted Mars capture, the lower dot indicates that the window would be wider, with the C3L lower and the Vhp limit of ≤ 7 km/s allowing a significantly wider window of approximately 10 days. Nevertheless, a 10 day launch window is considered operationally narrow and may be unacceptable for a real mission.

3.7.1.5 2011 and 2018 Missions. Similar analyses was performed on the 2011 and 2018 missions. We do not provide the details of the launch window analysis for the 2011 and 2018 missions in this report, but the resulting delta-V and parking orbit data are given in section 3.7.2.

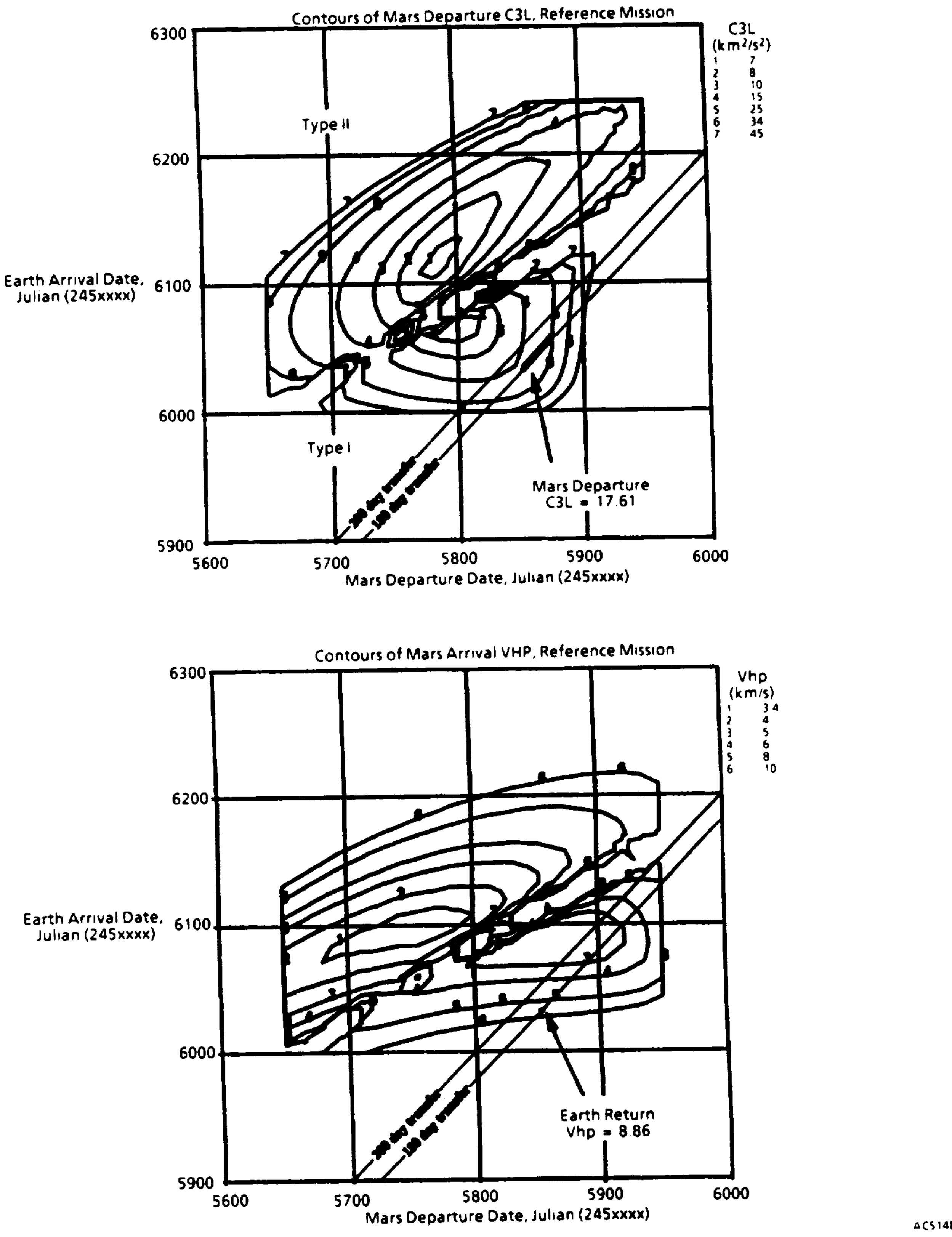

Figure 3-16. 2009 Mars Departure/Earth Arrival Windows

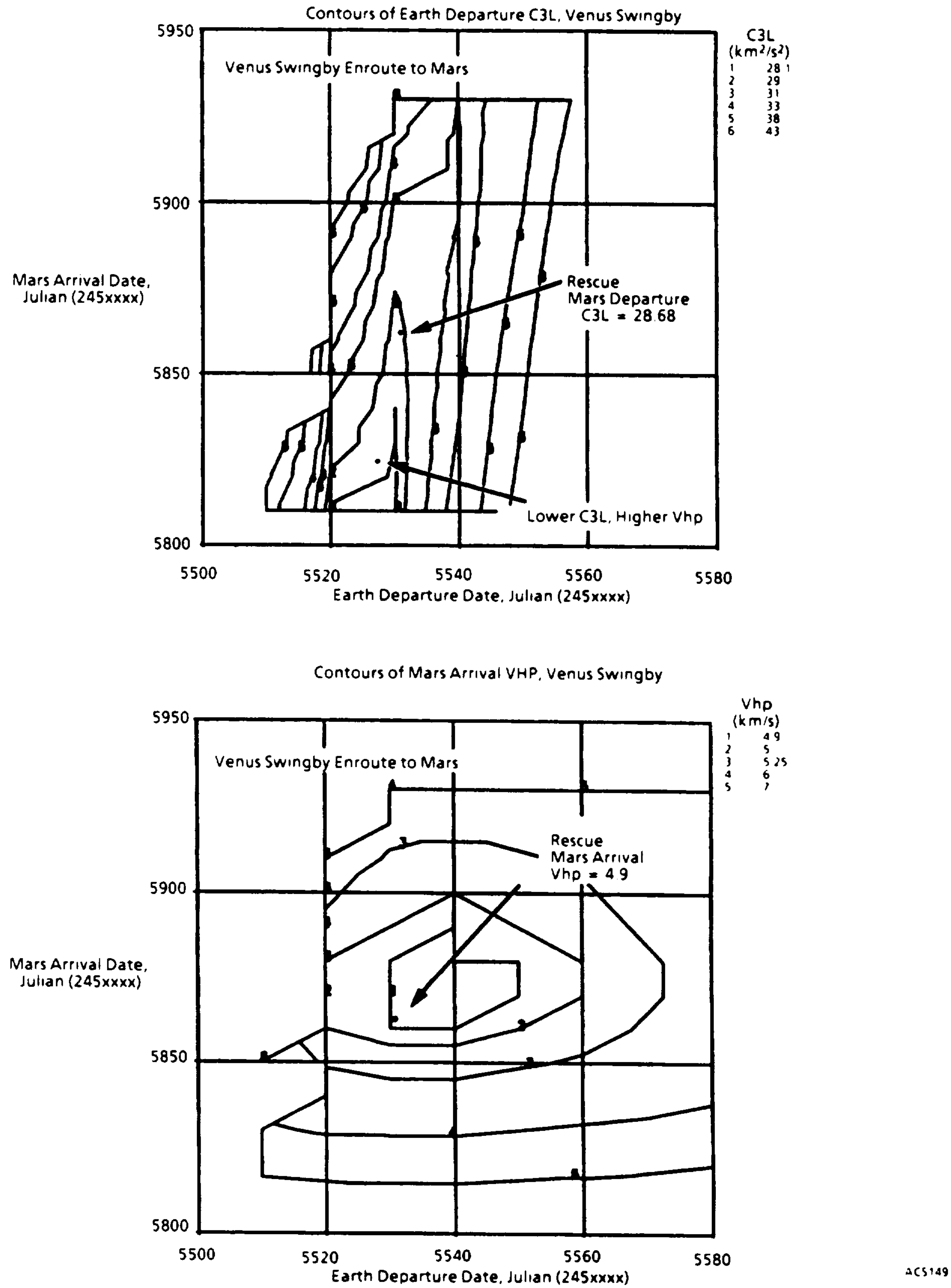

Figure 3-17. 2009 Rescue Windows with Venus Swingby

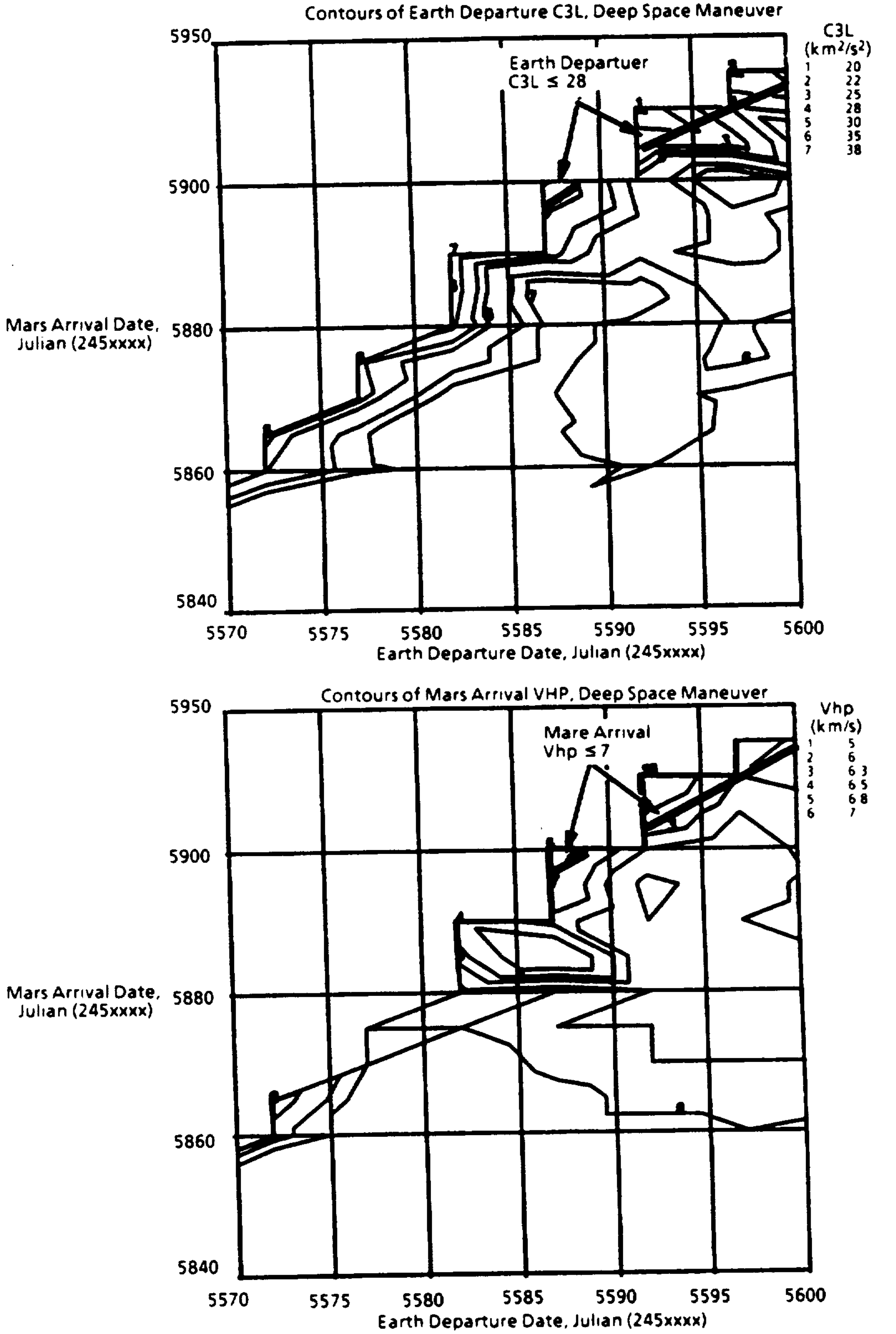

Figure 3-18. 2009 Rescue Windows with Deep Space Maneuver

3.7.2 Delta-V Budgets and Trajectory Data

A product of the mission and performance analysis is trajectory and delta-V data. This data is shown in figures 3-19 through 3-22 and can be used to compare the 2009 reference mission with follow-on 2011 and 2018 missions for nominal, minimum energy type I, and minimum energy type II trajectories.

Opportunity	Launch Date	Out (days)	Stay Time (days)	Return (days)	Mission Duration (days)	Earth C3L km^2/s^2	TMI* (m/s)	Mars Vhp (km/s)	Mars C3L km^2/s^2	Earth Vhp (km/s)	P O Delta-V (km/s)	Total Delta-V (m/s)
2009 Return Vehicle	9/13/07 2454356.5	344	1154	180	1678	13.43	4160	2.522	17.61	8.86	2780	6940
2009 Crew Aerobrake	10/30/09 2455134.5	180	N/A	N/A	180	20.09	4156	6.531	N/A	N/A	0	4156
2009 Crew Propulsive	11/19/09 2455154.5	180	N/A	N/A	180	29.36	4839	5.304	N/A	N/A	2610	7449
2011 Nominal	12/4/11 2455900.0	200	555	160	915	14.15	4191	5.400	27.05	6.90	5560	9751
2011 Type I minimum	11/14/11 2455880.0	245	447	248	940	9.79	3998	4.000	6.36	6.00	3360	7358
2011 Type II minimum	10/30/11 2455865.0	310	387	338	1025	9.59	3989	2.700	5.82	5.10	2620	6609
2018 Nominal	6/10/18 2458280.0	150	625	151	926	18.13	4364	3.500	22.18	3.20	3630	7994
2018 Type I minimum	5/6/18 2458245.0	196	565	191	952	7.96	3916	3.000	11.36	3.30	3260	7176
2018 Type II minimum	5/1/18 2458240.0	255	533	263	1051	8.98	3962	3.500	14.24	4.30	3250	7212

* TMI losses include g-loss of 300 m/s and plane change of 100 m/s

Figure 3-19. 2009 and 2011 Mission Trajectory Data

Opportunity	Launch Date	Out (days)	Stay Time (days)	Return (days)	Abort Duration (days)	Earth C3L km^2/s^2	TMI (m/s)	Mars Vhp (km/s)	Mars C3L km^2/s^2	Earth Vhp (km/s)	P O Delta-V (km/s)	Total Delta-V (m/s)
2009 VBS/Direct	12/11/10 2455532	328.43	30	280	638.5	28.68	4812	4.9	16.07	7.0	4730	9542
2009 (2011 Nominal)	21/4/11 2455900	200	555	160	915	14.15	4191	5.4	27.05	6.9	5560	9751
2009 Direct/DSM	12/4/11 2455900	200	30	294.8	524.8	13.83	4127	5.5	37.55	4.8	was not computed	was not computed
2011 Abort direct/VSB	1/17/14 2456675	175	100	280	555	13.91	4180	6.22	42.23	5.5	8390	11510
2011 (2014 Nominal)	1/17/14 2456675	175	512	230	917	13.71	4171	6.26	5.70	5.4	5120	9291
2018 Abort Direct/VSB	7/19/20 2459200	150	25	365	540	16.25	4278	4.50	27.69	4.5	4998	9276

* TMI losses include g-loss of 300 m/s and plane change of 100 m/s

Figure 3-20. Rescue Missions Trajectory Data

Opportunity	Inclination (deg)	Period (hours)	Periapsis Lighting (deg)	Periapsis Latitude (deg)	MOC* (m/s)	TEI (m/s)	AML (m/s)	P.O Delta-V (km/s)
2009 Return Vehicle	144	24.6	-16	36	890	1890	326	2780
2009 Crew Aerobrake	23.4	24.6	1.55	-4.8	0	N/A	N/A	0
2009 Crew Propulsive	20.7	24.6	5.2	-5.8	2610	N/A	N/A	2610
2011 Nominal	30	12.6	5.5	-27	2880	2680	66	5560
2011 Type I minimum	30	19.6	10	-28	1900	1460	418	3360
2011 Type II minimum	20	12.6	42	-18	1120	1500	582	2620
2018 Nominal	20	12.6	41	-14	1530	2100	139	3630
2018 Type I minimum	50	9.6	41	-30	1340	1920	436	3260
20198 Type II minimum	10	12.6	82	-8	1560	1690	39	3250

* g-losses for MOC, TEI are 50 m/s and 30 m/s respectively; TEI includes the AML

Figure 3-21. Parking Orbit Data for 2009, 2011, and 2018 Missions

Opportunity	Inclination (deg)	Period (hours)	Periapsis Lighting (deg)	Periapsis Latitude (deg)	MOC* (m/s)	TEI (m/s)	AML (m/s)	P.O Delta-V (km/s)
2011 Abort Direct/VSB in 2014	40	14.6	7.2	34	3460	3870	263	7330
2018 Abort Direct/VSB in 2020	10	24.6	4.8	2.6	2238	2760	201	4998

* g-losses for MOC, TEI are 50 m/s and 30 m/s respectively; TEI includes the AML

Figure 3-22. Parking Orbit Data for Selected Rescue Opportunities

General trajectory data are provided in figures 3-19 and 3-20. Nominal trajectory data for the reference mission, a nominal 2011 mission, and type I and type II comparison missions are given in figure 3-19. Trajectory data for a set of typical rescue mission are provided in figure 3-20. This data includes launch dates, outbound transfer time, Mars stay time, and return transfer time Included in these figure also are Earth/Mars departure C3L and arrival Vhp. Note that the TMI value include an estimated Earth departure g-loss and a computed 3-burn Earth departure plane change loss[7]. The parking orbit delta-V is the addition of MOC and TEI, including estimated g-losses and computed apsidal misalignment losses at Mars departure[8].

Mars elliptical parking orbit data are provided in figures 3-21 and 3-22. Parking orbit data for the reference 2009, a nominal 2011, and type I and type II 2011/2018 missions are shown in figure 3-21. For comparison purposes, parking orbit data for selected rescue missions are indicated in figure 3-22. These data of figures 3-21 and 3-21 are for optimal elliptical parking orbits that meet the operational constraint for manned missions of periapsis lighting angle ≥ 5 degrees. It can be seen from figure 3-21 that this constraint is not met, hence the 1.55 degrees light angle. The reason that the reference mission can not have adequate arrival lighting is based on the Sun/Mars lighting geometry at arrival. The part of the 2009 reference mission that places the return vehicle in Mars orbit is unmanned and does not descend to the surface, and therefore the negative lighting angle is irrelevant.

3.7.3 Failure Analysis

The mission/abort event trees used to model mission success and loss probabilities were more detailed than indicated by the figures presented earlier in this seciton, typically having about 30 events. This additional detail was needed to follow each abort sequence through to its end points, in order to obtain the probabiltiy distribution between mission success, mission loss, and crew loss. In order to avoid human error in the calculations (which are tedious), a computer code was developed with a convenient GUI to perform the calculations. The code was validated by repeating calculations of mission/abort event trees prepared earlier in the STCAEM contract, analyzed manually and carefully checked at that earlier time.

4.0 LUNAR SYNERGISM

4.1 INTRODUCTION

This investigation explored what could be done and what would be useful to do in a lunar test bed. The study was done in four main parts: a) a description of lunar evolution options that would permit testing of known required Mars major elements, b) a workup of the schedule of mission timing and launch requirements based on previous STCAEM work on First Lunar Outpost (FLO) and using a delivered launch capacity of 30 metric tonnes, c) a listing of elements and operations to be tested with a description of what the test objectives will be and an evaluation of the type of testing that could be accomplished within the time frame for incorporation into Mars mission systems and the expected resources available, d) an estimate of the impacts of lunar testing on a Mars mission schedule by using information on current and past programs for development and performance flows, and e) a trace of the heritage of the Mars Habitat through the FLO and past trade studies.

4.2 GENERAL APPROACH

This study started with the elements and operations from schemes outlined in the October 9,1992 NASA in-house Mars mission status report. As the elements were refined and developed they were incorporated into the lunar trial scenarios. For the full scenario development, which would dictate the timing of events, the FLO evolution was used as a starting program into which the Mars lunar testbed could be logically folded without interrupting lunar exploration and science. The items selected for lunar testing included specific elements of hardware and software, operations in the form of the ways to conduct tasks, the methodology to be used in conducting tasks and operations, and human interrelationships. All would directly benefit from a lunar trial as opposed to using a terrestrial test alone. That is; the items tested would derive some benefit from lunar testing that could not be achieved from a ground test alone. To this end, three different evolution schemes were generated that accomplished the trials at different times and with different operational intensities. All three use the 180 day trial as the arrival of the simulated Mars habitat (one tier elliptical structure) and it is from here that the chart showing the impact of a lunar habitat trial are taken for the later Mars mission impact study. Certain subsystems can and will be tested prior to the Mars habitat and were incorporated into the impact study timing.

4.3 DESCRIPTION OF OPTIONS

The three FLO derived evolution options investigated share this much in common: the mission uses four outpost missions of six crew to establish the base and build reserves for a 180-day mission that proceeds without a resupply flight and allows an on-orbit test of the Mars NTP or cryogenic engines. At the end of this time there is the possibility of moving the base or a part of it to a more desirable site as a result of the outpost exploration.

The "baseline" option, selected only as a starting point, continues with a 360 day mission that begins after initial restocking to allow a safety margin in case of an improper cargo delivery or other delay. After the Bio-chamber and a small lunar regolith oxygen plant for base atmosphere production arrives 135 days into the manned mission, there is a period of 225 days of crew isolation which receives no "packages from home" in the form of a cargo flight. This is roughly the duration of a Mars transit leg. The next mission in the baseline scenario is a 512 day mission that is an analog of the Mars surface durations expected, except that it will be resupplied every 90 days. The final manned mission shown in the baseline scenario is of undetermined length and begins the trial of a ground nuclear power supply.

The first alternative mission does not do the 360 day mission but goes directly into a 512 day mission. In this first alternative scenario the 512 day mission begins like the 360 day mission but continues to be supported by resupply landers. There is no extended isolation time without cargo contact.

The "umbilical cord" is still in place. The last noted manned mission is the one of undetermined length that tests the ground nuclear power supply. It has been moved forward as far as the currently known technology status will permit.

The second alternative also does not do the 360 day mission, but stockpiles supplies for the 512 day mission so the crew can spend the entire time in true isolation, without a resupply lander. This would mimic the isolation of a Mars surface stay for the first Mars manned mission and would be a fairly accurate test of systems and personnel together. It does require a number of unmanned cargo missions over a 2 year period during which time no lunar science or exploration takes place. Again, as in the first alternative, the ground nuclear power supply mission follows as soon as it appears the technology permits.

For each of these scenarios a timeline schedule and a tabulated flight manifest schedule are given. The flight manifest schedule shows the hardware, material, supplies brought, supplies on hand at the beginning of the mission, the type of spares brought and present at mission start, the stay time that can be supported at mission start and the launch centers between flights. Figures 4-1 and 4-2 are the timeline and tabulated schedule respectively for the "baseline" scenario. Figures 4-3 and 4-4 are the timeline and schedule for the first alternative and figures 4-5 and 4-6 are for the second alternative.

4.4 BENEFITS OF LUNAR TESTING AND TEST OBJECTIVES

A compilation of the elements of a Mars mission that could benefit from testing on the lunar surface was generated and is presented in figure 4-7. For each system, the subsystems that should undergo testing have been identified with an indication of needs in both the lunar and Mars environments. A commentary on the differences and needs for each subsystem segment is noted in the list of elements. This list of similarities and differences provided an insight into how much the lunar testing can help determine hardware, software, technique and interpersonal action modifications that must be instituted before committing the Mars mission first launch. From these items, specific tests and test objectives were established for analogs of the Mars systems or conditions for which lunar testing will provide data. These, together with the rationale of how the lunar test will impact the Mars equipment and operations are shown in figure 4-8. Specific listed items have some comment in the rationale statements on the criticality of hardware, software, system operations and tasks before and during the Mars mission to insure mission safety and success.

It should also be noted that we have not mounted a mission of this scale and complexity outside Earth orbit since December 1972, the last Apollo mission. The administrative and overall operations infrastructure are historical in nature. That is, many of the people and end to end institutions have been lost or diverted into other types of projects. These institutional frameworks and personnel expertise may be better redeveloped in a project that includes a near-Earth analog, that returns useful science, that will not totally usurp all resources and could be placed on hold for the main Mars effort if required.

4.5 SCHEDULE ANALYSES

Development schedules for identified critical long lead items were derived for comparison with each other to determine the critical elements with the greatest development time (long tent poles). They include the nuclear engine, ground nuclear power supply, the habitat, offloader and bio-chamber. These schedules comprise figures 4-9 and 4-10. Program elements were categorized as (a) common base or common development, (b) lunar hardware, and (c) Mars hardware. These are shown in the figures with different shadings as indicated on the key at the bottom. They include the lunar testing as a separate series of block time and allow for the results of these tests to

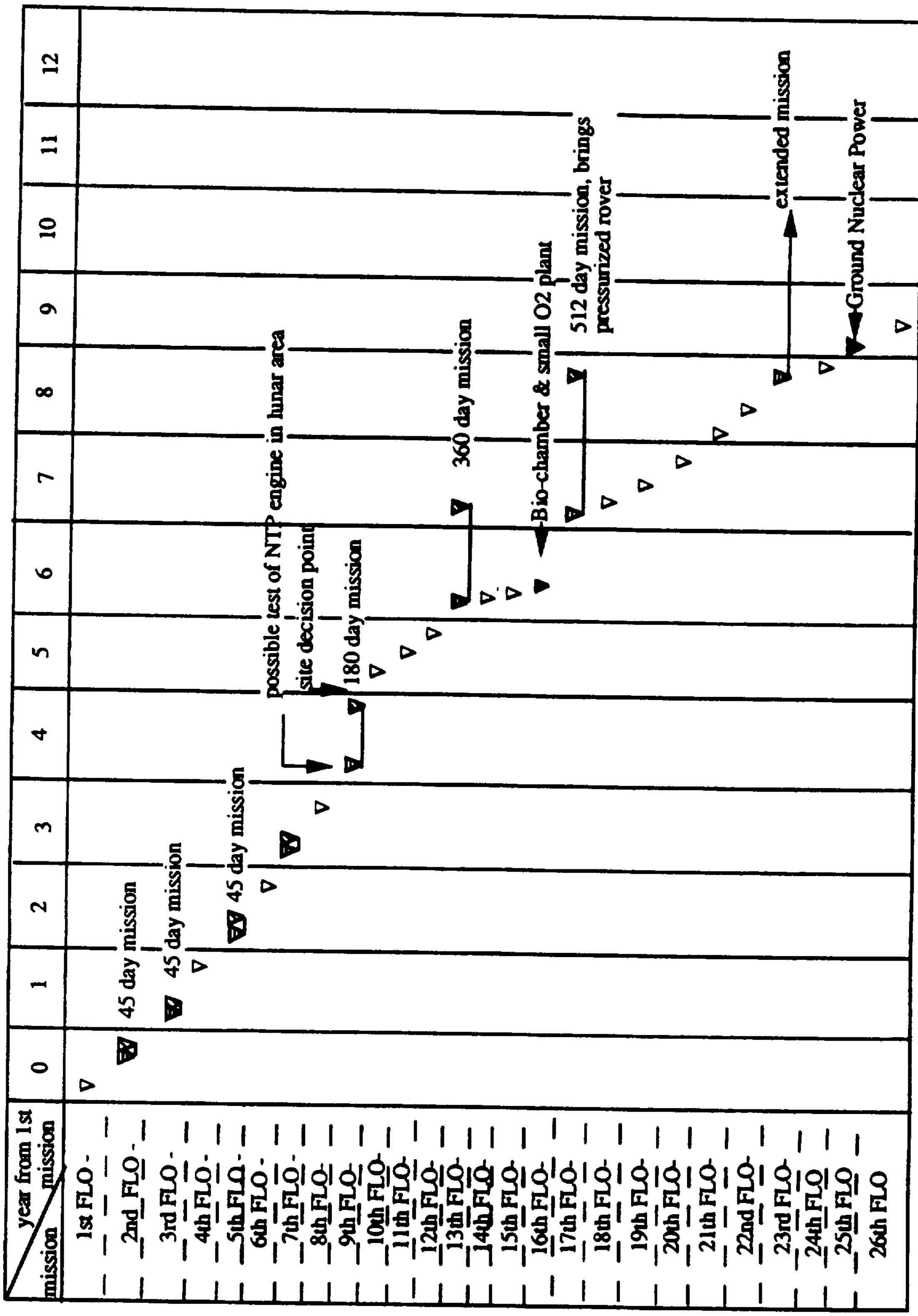

Figure 4-1. Baseline FLO Evolution Support of Mars Timeline

AC5151

Flight No.	Mission Type	Hardware brought	Material brought	Supplies brought	Supplies on surface	Spares brought	Spares on surface	Supported Staytime	Launch center
1	cargo	Outpost	Habitat on lander	one 45 terrestrial day stay	one 45 terrestrial day stay	contingency	contingency	one 45 terrestrial day stay	
2	manned	rover, science	rover	spares/ supply allocation?	one 45 terrestrial day stay	contingency	contingency	one 45 terrestrial day stay	6 months
3	manned	rover, science	rover	one 45 terrestrial day stay	one 45 terrestrial day stay	contingency	contingency	one 45 terrestrial day stay	6 months
4	cargo	LPLM with airlock	LPLM with airlock	one 45 terrestrial day stay	one 45 terrestrial day stay	critical initial	critical initial	one 45 terrestrial day stay	6 months
5	manned	science	spares/ supply allocation?	one 45 terrestrial day stay	@ mission start, two 45 day stays	critical initial +	critical initial +	one 45 day, extended stay or abort	6 months
6	cargo	surface Habitat, science	surface Habitat	one 45 terrestrial day stay	@ mission start, two 45 day stays	full initial +	full initial +	one 45 day, extended stay or abort	6 months
7	manned	science	spares/ supply allocation?	one 45 terrestrial day stay	@ mission start, two 45 day stays	full initial +	full initial +	one 45 day, extended stay or abort	6 months
8	cargo	#2 LPLM	#2 LPLM	one 45 terrestrial day stay	@ mission start, three 45 day stays	full initial +	full initial +	one 90 day, extended stay or abort	6 months
9	manned 180 day duration	science	spares/ supply allocation?	one 45 day, extended stay or abort	@ mission start, four 45 day stays	full initial +	full initial +	one 180 day, extended stay or abort	6 months

Figure 4-2. Baseline FLO Site Evolution

Flight No.	Mission Type	Hardware brought	Material brought	Supplies brought	Supplies on surface	Spares brought	Spares on surface	Supported Staytime	Launch center
10 begin 3 flights/ yr	cargo	LPLM #3 with airlock & side connection	LPLM #3 with airlock & side connection	one 90 terrestrial day stay*	none	as required	full initial +	one 90 terrestrial day stay**	120 days
11	cargo	surface Habitat #2	Surface Habitat #2	one 45 terrestrial day stay	135 terrestrial day stay @ start	as required	full initial +	one 135 terrestrial day stay**	120 days
12	cargo	LPLM #4 with airlock & side connection	LPLM #4 with airlock & side connection	one 90 terrestrial day stay	225 terrestrial day stay @ start	as required	full initial +	one 225 terrestrial day stay**	120 days
13 begin 4 flights/ yr	manned 360 day duration	science	science	one 45 terrestrial day stay	270 terrestrial day stay @ start	as required	full initial +	270 days	120 days
14 45 day centers	cargo	LPLM #5 with airlock & side connection	LPLM #5 with airlock & side connection	one 90 terrestrial day stay	360 @ cargo mission arrival	as required	full initial +	315 days	45 days
15	cargo	LPLM #6 with airlock & side connection	LPLM #6 with airlock & side connection	one 90 terrestrial day stay	360 @ cargo mission arrival	as required	full initial +	360 days	45 days
16	cargo	Bio-chamber & small oxygen plant	Bio-chamber & small oxygen plant	?	315 @ cargo mission arrival	as required	full initial +	315 days- (mission +)	45 days
17	manned 512 day duration	Pressurized rover	Pressurized rover	one 45 terrestrial day stay	90 @ mission arrival	as required	full initial +	135 days - end 360 day begin 512 day	225 days
18	cargo	LPLM #7 with airlock & side connection	LPLM #7 with airlock & side connection	one 90 terrestrial day stay	180 @ cargo mission arrival	as required	full initial +	180 days 467 days of mission left	90 days

* LPLM repacked with a reduction of spares and an increase in consumables
** Manned flights on hold pending decision on base placement and supply buildup
Note: some launches on 45 day centers and some on 90 day centers

Figure 4-2. Baseline FLO Site Evolution (Sheet 2)

Flight No.	Mission Type	Hardware brought	Material brought	Supplies brought	Supplies on surface	Spares brought	Spares on surface	Supported Staytime	Launch center
19	cargo	LPLM #8 with airlock & side connection	LPLM #8 with airlock & side connection	one 90 terrestrial day stay	180 @ cargo mission arrival	as required	full initial +	180 days 377 days left of mission	90 days
20	cargo	LPLM #9 with airlock & side connection	LPLM #9 with airlock & Side connection	one 90 terrestrial day stay	180 @ cargo mission arrival	as required	full initial +	180 days 289 days left of mission	90 days
21	cargo	LPLM #10 with airlock & side connection	LPLM #10 with airlock & side connection	one 90 terrestrial day stay	180 @ cargo mission arrival	as required	full initial +	180 days 197 days left of mission	90 days
22	cargo	LPLM #11 with airlock & side connection	LPLM #11 with airlock & side connection	one 90 terrestrial day stay	180 @ cargo mission arrival	as required	full initial +	180 days 107 days left of mission	90 days
23	manned*	science	science	one 45 terrestrial day stay	118 @ mission arrival	as required	full initial +	118 days of next mission (next cargo 45 days)	107 days
24	cargo	LPLM #12 with airlock & side connection	LPLM #12 with airlock & side connection	one 90 terrestrial day stay	163 @ mission arrival	as required	full initial +	163 days of stay covered	45 days
25	cargo	Ground Nuclear power reactor	Ground Nuclear power reactor	none	118 @ mission arrival	as required	full initial +	118 days of stay covered (next cargo 90 days)**	45 days
26	cargo	LPLM #13 with airlock & side connection	LPLM #13 with airlock & side connection	one 90 terrestrial day stay	118 @ mission arrival	as required	full initial +	135 days - end 360 day begin 512 day	90 days

* This manned mission arrives in 107 days
** 28 days of reserve consumables available if next 90 days resupply delayed

Figure 4-2. Baseline FLO Site Evolution (Sheet 3)

be incorporated into development, and possible alteration, of both Mars hardware and Mars operations techniques such as the resupply operations, laboratory analysis techniques, ground search and reconnaissance surface missions, dust control and other listed items. All of the development schedules were based on systems similar in configuration, employment of new technology, complexity and operations requirements for first item development, assembly, checkout and processing times. The primary information came from the Boeing Inertial Upper Stage (IUS) and Saturn vehicles. All development schedules are first cut approximations.

The first development schedule, figure 4-9 has not only the NTP engine and nuclear surface power schedules, but development lines for the then (April-May 1993) currently known and funded nuclear power/propulsion systems. This shows that if either the NTP or nuclear surface power programs require the "current programs" development, then there is likely to be a technology based impact on their development. The funding for these "current programs" has changed (decreased) since the time this comparison was made, making that situation, if it exists, worse. The NTP and nuclear surface power timelines are based on work previously done as part of the STCAEM trade studies for NTP (NTR) engine development and NEP/dual use power/ground power reactor systems development known at that time. This information has been updated as far as design

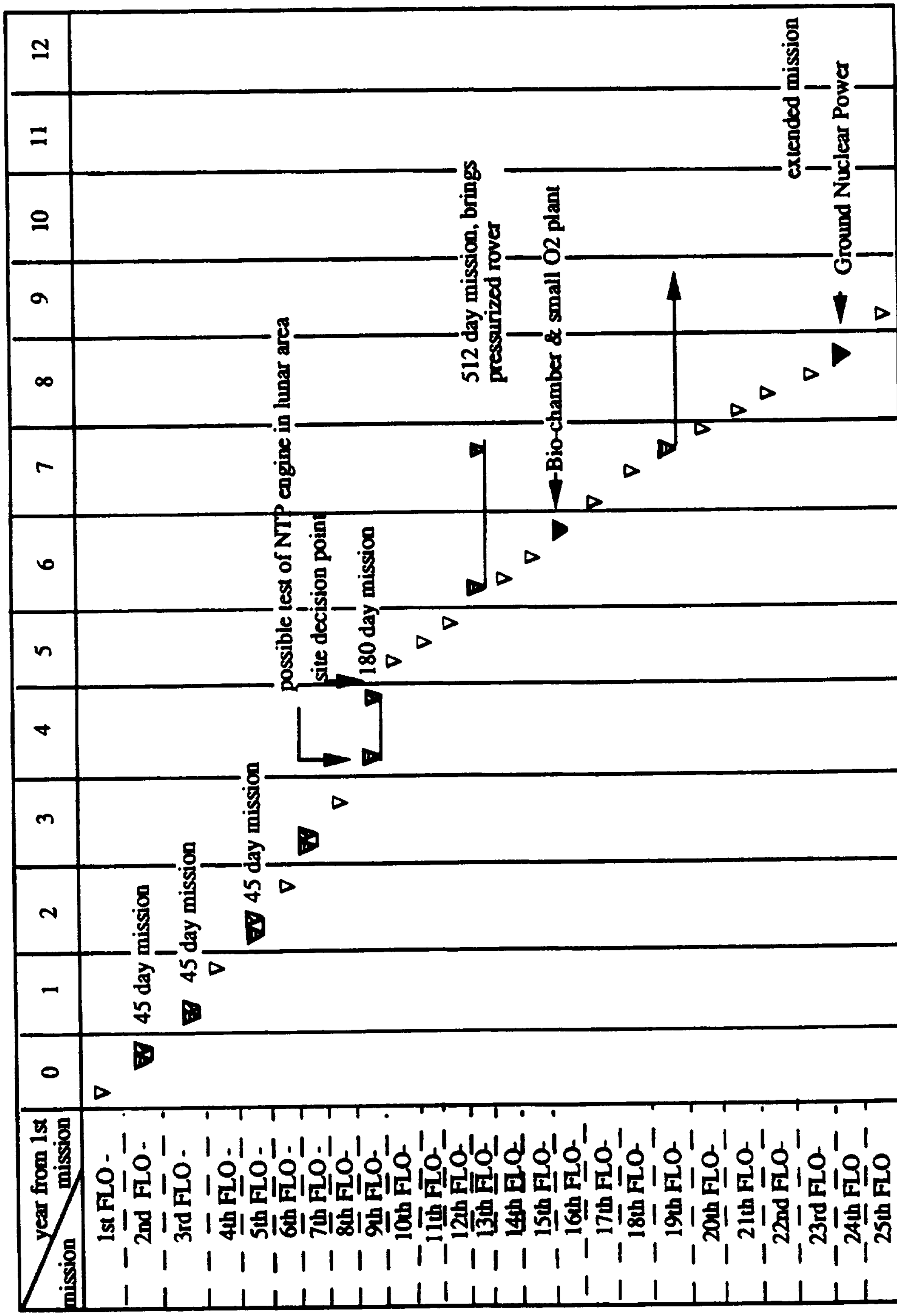

Figure 4-3. FLO Evolution Support of Mars Timeline - First Alternative

ACS152

Flight No.	Mission Type	Hardware brought	Material brought	Supplies brought	Supplies on surface	Spares brought	Spares on surface	Supported Staytime	Launch center
1	cargo	Outpost	Habitat on lander	one 45 terrestrial day stay	one 45 terrestrial day stay	contingency	contingency	one 45 terrestrial day stay	
2	manned	rover, science	rover	spares/ supply allocation?	one 45 terrestrial day stay	contingency	contingency	one 45 terrestrial day stay	6 months
3	manned	rover, science	rover	one 45 terrestrial day stay	one 45 terrestrial day stay	contingency	contingency	one 45 terrestrial day stay	6 months
4	cargo	LPLM with airlock	LPLM with airlock	one 45 terrestrial day stay	one 45 terrestrial day stay	critical initial	critical initial	one 45 terrestrial day stay	6 months
5	manned	science	spares/ supply allocation?	one 45 terrestrial day stay	@ mission start, two 45 day stays	critical initial +	critical initial +	one 45 day, extended stay or abort	6 months
6	cargo	surface Habitat, science	surface Habitat	one 45 terrestrial day stay	@ mission start, two 45 day stays	full initial +	full initial +	one 45 day, extended stay or abort	6 months
7	manned	science	spares/ supply allocation?	one 45 terrestrial day stay	@ mission start, two 45 day stays	full initial +	full initial +	one 45 day, extended stay or abort	6 months
8	cargo	#2 LPLM	#2 LPLM	one 45 terrestrial day stay	@ mission start, three 45 day stays	full initial +	full initial +	one 90 day, extended stay or abort	6 months
9	manned 180 day duration	science	spares/ supply allocation?	one 45 day, extended stay or abort	@ mission start, four 45 day stays	full initial +	full initial +	one 180 day, extended stay or abort	6 months

Figure 4-4. FLO Site Evolution - First Alternative

Flight No.	Mission Type	Hardware brought	Material brought	Supplies brought	Supplies on surface	Spares brought	Spares on surface	Supported Staytime	Launch center
10 begin 3 flights/ yr	cargo	LPLM #3 with airlock & side connection	LPLM #3 with airlock & side connection	one 90 terrestrial day stay*	none	as required	full initial +	one 90 terrestrial day stay**	120 days
11	cargo	surface Habitat #2	Surface Habitat #2	one 45 terrestrial day stay	135 terrestrial day stay @ start	as required	full initial +	one 135 terrestrial day stay**	120 days
12	cargo	LPLM #4 with airlock & side connection	LPLM #4 with airlock & side connection	one 90 terrestrial day stay	225 terrestrial day stay @ start	as required	full initial +	one 225 terrestrial day stay**	120 days
13 begin 4 flights/ yr	manned 512 day duration	pressurized rover	pressurized rover	one 45 terrestrial day stay	270 terrestrial day stay @ start	as required	full initial +	270 days	120 days
14	cargo	LPLM #5 with airlock & side connection	LPLM #5 with airlock & side connection	one 90 terrestrial day stay	270 @ cargo mission arrival	as required	full initial +	270 days 422 days of mission left	90 days
15	cargo	LPLM #6 with airlock & side connection	LPLM #6 with airlock & side connection	one 90 terrestrial day stay	270 @ cargo mission arrival	as required	full initial +	270 days 322 days of mission left	90 days
16	cargo	Bio-chamber & small oxygen plant	Bio-chamber & small oxygen plant	?	180 @ cargo mission arrival	as required	full initial +	180 days - 242 days of mission left	90 days
17	cargo	LPLM #7 with airlock & side connection	LPLM #7 with airlock & side connection	one 90 terrestrial day stay	180 @ mission arrival	as required	full initial +	180 days - 152 days of mission left	90 days
18	cargo	LPLM #8 with airlock & side connection	LPLM #8 with airlock & side connection	one 90 terrestrial day stay	180 @ cargo mission arrival	as required	full initial +	180 days 62 days of mission left	90 days

* LPLM repacked with a reduction of spares and an increase in consumables
** Manned flights on hold pending decision on base placement and supply buildup

Figure 4-4. FLO Site Evolution - First Alternative (Sheet 2)

Flight No.	Mission Type	Hardware brought	Material brought	Supplies brought	Supplies on surface	Spares brought	Spares on surface	Supported Staytime	Launch center
19	manned* begin next mission	science	science	one 45 terrestrial day stay	163 @ cargo mission arrival	as required	full initial +	163 days end 512 day mission	62 days
20	cargo	LPLM #9 with airlock & Side connection	LPLM #9 with airlock & Side connection	one 90 terrestrial day stay	163 @ cargo mission arrival	as required	full initial +	163 days	90 days
21	cargo	LPLM #10 with airlock & side connection	LPLM #10 with airlock & side connection	one 90 terrestrial day stay	163 @ cargo mission arrival	as required	full initial +	163 days	90 days
22	cargo	LPLM #11 with airlock & side connection	LPLM #11 with airlock & side connection	one 90 terrestrial day stay	163 @ cargo mission arrival	as required	full initial +	163 days	90 days
23	cargo	LPLM #12 with airlock & side connection	LPLM #12 with airlock & side connection	one 90 terrestrial day stay	163 @ mission arrival	as required	full initial +	163 days	107 days
24	cargo	Ground Nuclear power reactor	Ground Nuclear power reactor	none	73 @ mission arrival	as required	full initial +	73 days** of stay covered (next cargo 45 days)	45 days
25	cargo	LPLM #13 with airlock & side connection	LPLM #13 with airlock & side connection	one 90 terrestrial day stay	118 @ mission arrival	as required	full initial +	118 days	45 days

* This manned mission arrives in 62 days
** early resupply of 90 day consumables required

Figure 4-4. FLO Site Evolution - First Alternative(Sheet 3)

fabrication and testing schedules based on the assumption that work has continued at various places such as NASA Lewis Research Center and that many concepts and designs are still being explored. The process then requires investigation and selection of one of these options, reducing some of the design, development and testing times.

Figure 4-10 contains the development schedules for the habitat, the module offloader and the bio-chamber. All these schedules include a lunar development branch. Of these the bio-chamber takes the longest time, but while it was the longest, the first mission could be performed without it. The offloader, which will be needed, while it is designed, developed and tested in conjunction with the habitat, it is not required for the first manned lunar mission, and lags slightly in the need date. It has a development schedule shorter than the habitat and it is not needed as early, leaving the habitat as the longest critical-path element for the first manned Mars mission.

The lunar impact schedules, figure 4-11 and 4-12, compare the development schedule of the NTP engine and the habitat with and without a lunar testing development branch. In the case of both systems, but particularly the habitat, some time is spent in development producing a system for which operations will be analogous in both the lunar and Mars environments, i. e. a lunar system similar enough to provide feedback data to a Mars design. The "price" is time for doing this that can be subtracted from the development schedule of a design that works strictly for the Mars environment. Hence

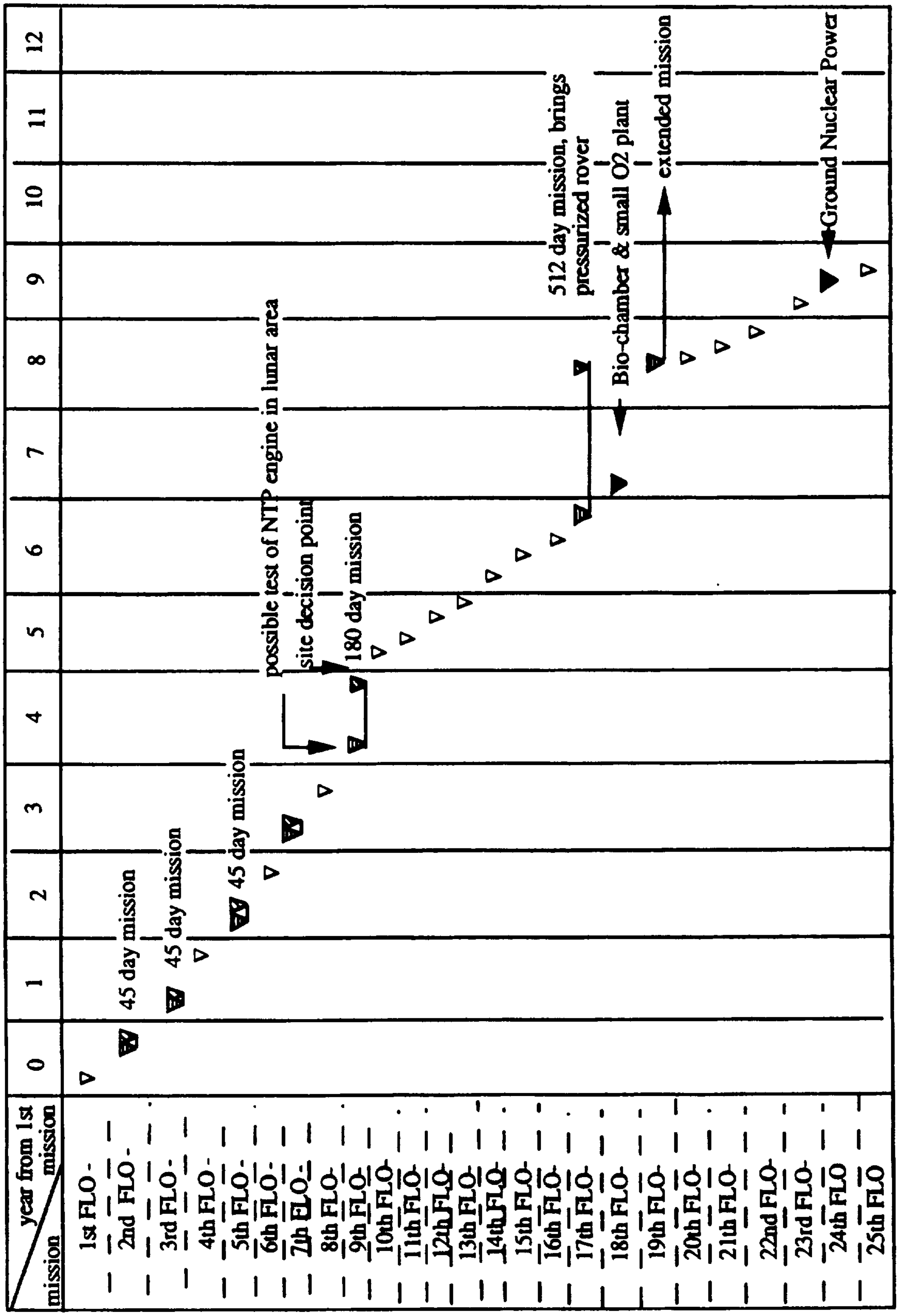

Figure 4-5. FLO Evolution Support of Mars Timeline - Second Alternative

AC5153

Flight No.	Mission Type	Hardware brought	Material brought	Supplies brought	Supplies on surface	Spares brought	Spares on surface	Supported Staytime	Launch center
1	cargo	Outpost	Habitat on lander	one 45 terrestrial day stay	one 45 terrestrial day stay	contingency	contingency	one 45 terrestrial day stay	
2	manned	rover, science	rover	spares/ supply allocation?	one 45 terrestrial day stay	contingency	contingency	one 45 terrestrial day stay	6 months
3	manned	rover, science	rover	one 45 terrestrial day stay	one 45 terrestrial day stay	contingency	contingency	one 45 terrestrial day stay	6 months
4	cargo	LPLM with airlock	LPLM with airlock	one 45 terrestrial day stay	one 45 terrestrial day stay	critical initial	critical initial	one 45 terrestrial day stay	6 months
5	manned	science	spares/ supply allocation?	one 45 terrestrial day stay	@ mission start, two 45 day stays	critical initial +	critical initial +	one 45 day, extended stay or abort	6 months
6	cargo	surface Habitat, science	surface Habitat	one 45 terrestrial day stay	@ mission start, two 45 day stays	full initial +	full initial +	one 45 day, extended stay or abort	6 months
7	manned	science	spares/ supply allocation?	one 45 terrestrial day stay	@ mission start, two 45 day stays	full initial +	full initial +	one 45 day, extended stay or abort	6 months
8	cargo	#2 LPLM	#2 LPLM	one 45 terrestrial day stay	@ mission start, three 45 day stays	full initial +	full initial +	one 90 day, extended stay or abort	6 months
9	manned 180 day duration	science	spares/ supply allocation?	one 45 day, extended stay or abort	@ mission start, four 45 day stays	full initial +	full initial +	one 180 day, extended stay or abort	6 months

Figure 4-6. FLO Site Evolution - Second Alternative

Flight No.	Mission Type	Hardware brought	Material brought	Supplies brought	Supplies on surface	Spares brought	Spares on surface	Supported Staytime	Launch center
10 begin 4 flights/ yr	cargo	LPLM #3 with airlock & side connection	LPLM #3 with airlock & side connection	one 90 terrestrial day stay*	none	as required	full initial +	one 90 terrestrial day stay**	90 days
11	cargo	surface Habitat #2	Surface Habitat #2	one 45 terrestrial day stay	135 terrestrial day stay @ start	as required	full initial +	one 135 terrestrial day stay**	90 days
12	cargo	LPLM #4 with airlock & side connection	LPLM #4 with airlock & side connection	one 90 terrestrial day stay	225 terrestrial day stay @ start	as required	full initial +	one 225 terrestrial day stay**	90 days
13	cargo	LPLM #5 with airlock & side connection	LPLM #5 with airlock & side connection	one 90 terrestrial day stay	315 terrestrial day stay @ start	as required	full initial +	315 days	90 days
14	cargo	LPLM #6 with airlock & side connection	LPLM #6 with airlock & side connection	one 90 terrestrial day stay	405 @ cargo mission arrival	as required	full initial +	405 days	90 days
15	cargo	LPLM #7 with airlock & side connection	LPLM #7 with airlock & side connection	one 90 terrestrial day stay	495 @ cargo mission arrival	as required	full initial +	495 days	90 days
16	cargo	LPLM #8 with airlock & side connection	LPLM #8 with airlock & side connection	one 90 terrestrial day stay	585 @ cargo mission arrival	as required	full initial +	585 days	90 days
17	manned 512 day duration	pressurized rover	pressurized rover	one 45 terrestrial day stay	630 terrestrial day stay @ start	as required	full initial +	630 days	90 days
18	cargo	Bio-chamber & small oxygen plant	Bio-chamber & small oxygen plant	?	540 @ cargo mission arrival	as required	full initial +		90 days

* LPLM repacked with a reduction of spares and an increase in consumables
** Manned flights on hold pending decision on base placement and supply buildup

Figure 4-6. FLO Site Evolution - Second Alternative (Sheet 2)

Flight No.	Mission Type	Hardware brought	Material brought	Supplies brought	Supplies on surface	Spares brought	Spares on surface	Supported Staytime	Launch center
19	manned* begin next mission	science	science	one 45 terrestrial day stay	73 @ cargo mission arrival	as required	full initial +	73 days** end 512 day mission	512 days
20	cargo	LPLM #9 with airlock & side connection	LPLM #9 with airlock & Side connection	one 90 terrestrial day stay	118 @ cargo mission arrival	as required	full initial +	118 days	45 days
21	cargo	LPLM #10 with airlock & side connection	LPLM #10 with airlock & side connection	one 90 terrestrial day stay	163 @ cargo mission arrival	as required	full initial +	163 days	45 days
22	cargo	LPLM #11 with airlock & side connection	LPLM #11 with airlock & side connection	one 90 terrestrial day stay	163 @ cargo mission arrival	as required	full initial +	163 days	90 days
23	cargo	LPLM #12 with airlock & side connection	LPLM #12 with airlock & side connection	one 90 terrestrial day stay	163 @ mission arrival	as required	full initial +	163 days	90 days
24	cargo	Ground Nuclear power reactor	Ground Nuclear power reactor	none	73 @ mission arrival	as required	full initial +	73 days (next cargo mission in 45 days)	90 days
25	cargo	LPLM #13 with airlock & side connection	LPLM #13 with airlock & side connection	one 90 terrestrial day stay	118 @ mission arrival	as required	full initial +	118 days	45 days

* This manned mission arrives in 62 days
** early resupply of 90 day consumables required

Figure 4-6. FLO Site Evolution - Second Alternative(Sheet 3)

the difference in development and initial checkout time for the Mars mission alone. Using this as a point of departure for reducing the Mars-alone schedules as much as possible for the required longest lead critical elements, it appears that the addition of lunar testing adds four years to each of the schedules.

4.6 HABITAT HERITAGE

The Mars Habitat used in the NASA in-house study is a three-tiered extended elliptical structure, based in part, on a single tier (one deck) lunar configuration studied as an alternative habitat structure for the First Lunar Outpost. This lunar configuration, in turn, had its roots in the analyses presented in Long-Duration Habitat Study of March 1990 which was done as part of this contract. In that study a matrix of 5 crew sizes, three module sizes and 6 diameters was analyzed under varying conditions of gravity, orientation, topology and structure. From these data, 1480 distinct options were generated of which 150 concepts were focused on as likely candidates, including the elliptical and extended elliptical family of configurations. The lunar elliptical habitat, derived as an alternate FLO configuration, is shown in figure 4-13 with a volume analysis given in figure 4-14. More information on this particular lunar configuration can be found in the FLO Alternatives section of the Space Transportation and Analysis for Exploration Missions Phase Three Final Report, June 1993, D615-10062-2.

System	Subsystem	Lunar	Mars	Comments
All Surface Modules	Mobile Cradle	• works in 1/6 g • telepersent to start, but may evolve to autonomous • power is continuous during the long lunar day, ∴ mobility, service systems & recharging are continuous during this time	• works in 1/3 g • may be autonomous to start • power is not continuous, unless an RTG or other source than solar used. Solar cell arrangement will be larger and heavier.	• use must be verified on Moon before Mars operations • rapid power shifts during the Mars day & night, must be accounted for, if the system is not to be degraded rapidly and have mobility move in "starts & fits" and may have fatigue failure problems
	Surface Connection and Docking mechanism	• connection techniques done man-assisted, then autonomous	• connections must be autonomous & done to minimize dust intrusion	• must be verified on the Moon before Mars operations
	Dust control	• at all mechanical interface surfaces & electronics	• aeolian driven dust problem	• techniques must be verified on the Moon before Mars operations
	Resupply	• supply and resupply techniques established in lunar operation	• supply techniques involve larger quantities of material and retrieval of greater/larger? items	• resupply techniques may have to be modified for Mars
Lander/ Ascent Vehicle	Precision Landing	• needs repeated landings at one area (1 km circle)	• needs repeated landings at one area (? circle) from multiple orbits	landing for Mars will be more difficult, as the capture orbits will be different for each opportunity, then entry must be to the same point
	Offloading Mechanism	• Offloading of major elements must be verified	offloading of major elements must be conducted in high dust conditions	the difference in gravity and possible configuration differences in the modules will require a stronger and possibly more elaborate mechanism
	Mechanical portion of vehicle health management	• verification of dormant engine operability surface and possibility remote on orbit	• verification of dormant engine operability in both surface and remote on orbit critical	verification on on orbit engine operations critical to abort to surface, once the ascent is accomplished, return to Mars base may not be possible.
Habitat	Laboratory portion			
	• ISRU	• experiments in extraction and analysis processes O_2 extraction may be used for atmospheric makeup	• experiments in extraction and analysis processes	ISRU experiments will differ in direction, lunar will look for O_2, H_2 and material use; Mars will look for H_2O, ices, biological evidence and material use
	• Medical	• physiological monitoring in reduced gravity • physiological monitoring of crew in long duration confined conditions	• physiological monitoring in reduced gravity • physiological monitoring of crew in long duration confined conditions	Mars physiological evaluation will be on speed of recovery from microgravity conditions to partial gravity and detection of any extraterrestrial pathogens
	• Geophysical	• sensors, field observations, etc.	• sensors, field observations, etc.	Sensors and systems will have to be modified from the lunar equipment to Mars atmosphere and dust conditions
	Life Support Systems	First in-situ tests for long durations	Long duration LSS critical to mission	Common systems under common conditions
Pressurized Rover		May or may not be used/tested in the lunar environment, if it is, it will come as a science payload of 5t	It is established as part of the initial landed cargo	for lunar operations may require a single launch, a manned lander with only the pressurized rover & consumables

Figure 4-7. Common Lunar - Mars Elements

System	Subsystem	Lunar	Mars	Comments
Bio-chamber	• Plant Growth	• 24 hr. night-day cycle maintained during lunar day & night • Preliminary plant selections for growth in reduced gravity • a "long-day" cycle similar to an Earth polar summer might be tried • automation techniques or dormant ecologies should first be tested here	• 24 hr. night-day cycle maintained during the Martian cycle or the Martian day & night might be tested for use • plant selections may be refined, for those that can tolerate a dormant time period, or methods developed to place the Bio-chamber in a dormant state for between missions	Plant growth in Mars soil trials will involve either an external area isolated from the Mars surface and the Base atmosphere/systems or a section of the Bio-chamber that is isolated from crew and Earth-based operating systems (prevention of cross-contamination between crew, other plant experiments and the Mars soil experiments). These techniques need to be provided in the lunar Bio-chamber trials.
	• Bio-decontamination	• techniques for not cross contaminating the bio-chamber with fungi, bacteria, etc., from the crew or the selection of uncontaminated/tolerant plants done here	• a technique for testing growth in the Martian soil, without infection from or to the soil samples may be don in-situ	
Software	Terrain traverse program:			
	• Mobility actuation	Lunar terrain and soil are known by Apollo work, but work in automation still required	Mars soil consistency and local terrain are not well known	Soil characteristics will vary widely between lunar an Mars surfaces (presence of moisture, consistency of dust, resultant wear etc.). This must be accounted for in system preparation.
	• Visual Interpretation	Lunar sharp contrasts are known from Apollo work, automation required	Mie scattering of light may cause different perceptions	Visual perceptions will change between the Moon and Mars, due to atmosphere presence or absence, reflection, dust, etc. affecting both the depth perception, and ground route identification
	• Route identification program	Apollo data and Lunar Mapper will give a more detailed idea of landmarks	Mars is not well known; not in specific details. Mars Rover or Sample Return may improve this	
	IVHM:			
	• cold engine start verification	Will allow a lead time for a rescue lander to be sent while supplies are available	Will be critical for both ascent and on-orbit engines as abort-to-surface is primary option	This is more of a concern for Mars, where there is a long duration between onset of the problem and rescue capability
	• Habitat verification & dormancy	The ability to "wake up" & interrogate the systems prior to mission commit then "mothball" them add to architecture flexibility & extend system life	Habitat condition must be known prior to orbit insert	Remote verification of the habitat conditions can save a mission by knowing needed repairs prior to mission commit
	• Bio-chamber isolation & dormancy	Operational testing ground for Mars systems	For a long duration stay the operation must be isolated and dormancy/sealing verified between missions	Cross contamination could be critical in both isolated environments. The methods must be provided prior to Mars missions
Software	Data Management & System Prioritization	TBD	TBD	Critical systems & monitoring will have some specific aspects to the Moon or Mars due to the time for resupply/rescue. For lunar outpost/base priorities will change with base expansion
NTP/Cryo	Engine/propulsion system	provides an opportunity to recover and evaluate an engine that has been in space for a significant time and perhaps repeatedly fired		An in-space dormancy period with/without restart, can be recovered for physical examination on cool-down

Figure 4-7. Common Lunar - Mars Elements (Continued)

System	Subsystem	Lunar	Mars	Comments
All Surface Modules	Mobile cradle	verify: • operation of cradle supports thorough offload, transverse & module connections	• cradle system must be used on Mars for cargo transport to form the base	• without a lunar trial, an increased risk of a malfunction@ Mars or increased mass to insure cradle function
		• operation of cradle suspension through offload, transverse & module connections	• comparison of suspension action in 1g and 1/6g will allow more accurate estimate of 1/3g needs	• suspension trial similar risks to cradle operation, potential failure or increased mass
		• operation of cradle wheels through offload, transverse & module connections	• a lunar trial of wheels that support muli-tonne system over ≥ 1km will help determine wheel life per distance traveled	• here a lunar trial is critical, wheel failure may determine the distance from the base area that the cargo must land
		• structural support on tie-downs • thermal insulation through tie-downs	• established tie-down and insulation techniques for shift and slump conditions in long duration	• a lunar trial will help determine heat flow and support compensation under temperature extremes, useful since the Mars permafrost layer is unknown and may effect hab. positioning & stability
	Surface Connection and Docking Mech.	• verify methods of physical connection • establish methods of resource cross-connections (manned)	• cargo components must be connected on Mars surface with pressurized (habitable) connections	• without a long duration trial, in vacuum with reduced gravity and after transport, an increased risk of physical pressurized connections failing
	Dust Control	• establish methods of dust exclusion on regular, repeated operations & connections	• dust control for contamination due both to human activity, landing, & aeolian activity will effect long term habitability	• here will be a critical test of dust contamination control on entering and exiting the habitable volume, check of dust problem due to landing
	Resupply	• establish methods & techniques of manual and log module resupply physical requirements	• physical resupply will be done @ Mars from the onset, manual resupply will give insight to crew endurance, log module will show inventory req.	• without "real situation" endurance trials, long duration physical effects may not be determined before Mars launch and/or log module requirements not known
Lander/ Ascent Vehicle	Precision Landing	• verify point to point navigation accuracy • test use of beacons for guidance • estimate CEP size (footprint) for landing	• accuracy in landing from several orbit inclinations are required • beacon guidance may be the only method for "pinpoint" landing • acceptable limit for manned transport, manned logistics transfer & landing dust can be determined	• for all precision landing questions, the methods and limits must be determined prior to the first Mars cargo launch. This is a question not only of what terrain is acceptable and where it is, but how to access it and the physical tolerances of the hardware, software and people. These questions can be determined by a lunar trial, which a close approximation that allows change & retest
	Offloading Mechanism	• check of technique - stability - terrain tolerance - disconnects (disconnect actions & loses)	• similar offloading techniques will be used for Mars cargo & manned flights	• with no lunar trial, increased risk of either offloading hangup @ Mars or increased mass to insure offload function
	mechanical portion of vehicle health management	• checks of individual systems response time (remote & manned) • system activation in test with reduced gravity	• physical check of how far the system can be excited and how long it takes to check the system and know the response is correct	• This is a critical check of the vehicle health management system. The physical change in reduced gravity will effect both the activation level & response time. If not conducted on the Moon, the degree of change may not be accounted for

Figure 4-8. Lunar - Mars Testing Elements

System	Subsystem	Lunar	Mars	Comments
Habitat	Laboratory portion • ISRU	• establish methods of IRSU tests and waste disposal • establish ISRU tool usage IVA & EVA	• methods may be critical to Mars O_2 & fuel production • direct analog to Mars requirements	• if the systems and techniques are not tested on the Moon, then cross-containment problems that do not show in Earth or SSF tests may develop @ Mars
	• Medical	• establish long duration physical health monitoring needs of crew • verify tool requirements/ usefulness for crew health • establish long duration psychological health monitoring needs of the crew	• Mars will have twice the gravity, but physical operations and mental stress in confined areas on lunar trial for an extended period will test the physical and psychological limits that can not be done on Earth. Tools, techniques and countermeasures can be identified & developed	• Earth testing alone cannot reproduce the "on your own" feelings and inventiveness that the isolation of a lunar trial can produce (even overwintering in Antarctica) especially with the added environmental considerations
	• Geophysical	• verify sensor systems & field observation tech.	• direct analog to Mars requirements	• shadows, lighting differences sensor outputs must be recreated on Earth
	Life Support Systems	• check of all common systems under "real service" conditions	• long duration lunar trial is the best Mars analog; more frequent in and out stresses than SSF	• Earth check would have to be run in a large man-rated vacuum system ($) for a long time ($$) with human ingress/egress ($$$)
Pressurized Rover		• check of all common systems under "real service" conditions	• long duration lunar trial is the best Mars analog; more frequent use stresses under vacuum & dust than earth	• in a terrestrial vacuum mobility may not be checked at the same time, with any amount of dust
Bio-chamber	• Plant Growth	• test plant selections • establish plant dormancy/production cycle • verify plant material storage technique: - crop storage - waste product (store/compost) • verify adequate support material provided	• plant selection for reduced gravity & crew health must be selected prior to Mars missions Plant life cycles in reduced gravity must be established as well as crop rotation, methods of waste disposal (bacteriological content, potential crop and crew contamination) and proper crop storage with isolation determined	once committed to a Mars mission there will be no chance to change the crops or "freshen" the system, either by removal and decontaminating the plant waste products, the addition of new plant stock or changing out the soil and water systems. This is an area of potentially serious biological cross-contamination
	• Bio-Decontamination	• verify seals between bio-chamber/living and working quarters function • verify isolation & decontamination system for : - water - air - humidity - waste products	• the same system that is used in the lunar trials will be used in the Mars mission system • the same system that is used in the lunar trials will be used in the Mars mission system	• with reference to the above plant growth & crew health needs these systems will be more critical on Mars where a problem must be dealt with in place and where the potential for not only crew contaminating the plants but the Martian environment through soil contact may pose a threat
Software	Terrain traverse program: • Mobility actuation • Visual interpretation • Route identification program	• verify both autonomous and manned control function for the habitat/log modules • verify the ability to distinguish ground hazards in all lighting conditions • verify the ability to track a beacon around obstacles, follow tracks and route markers	• cargo delivered hab., bio-chamber and logistics carriers will have to move to the base area, with the crew performing the inter-connections • visual conditions may or may not be better on Mars, but a system working here will work on Mars • several forms of route identification & can be tried. Several forms of regolith/sand firmness testing can be applied	• lunar trial will help establish the terrain limits, the extent of mobility to the base that can be done by machines alone and the amount needing human guidance, the requirements for computer capability, weather beacons help, and how much "base gathering" can be done prior to crew arrival

Figure 4-8. Lunar - Mars Testing Elements (Continued)

System	Subsystem	Lunar	Mars	Comments
Software (Continued)	IVHM			
	• cold engine start verification	• test, develop & verify ability to determine engine status without engaging engine thrust	• critical for Mars mission crew safety, as it may determine when to terminate the mission	• lunar conditions permit a real situation test of engine component limits under reduced gravity & dust laden vacuum
	• Habitat verification & dormancy	• test, checkout and verify ability to status and "mothball" hab.	• critical for Mars mission safety, as it may determine when to terminate the mission	• lunar program offers a real situation, repeated use test that will define realistic requirements prior to Mars commit
	• Bio-chamber isolation & dormancy	• verify contamination detection/alert/ isolation & recover program • verify nominal "moth-balling" program	• critical for Mars mission as Bio-chamber may have to produce part of the food supply in an abort situation	• isolation and reduced gravity can't be reproduced on Earth, in an earth test contamination may come from outside sources
Software	Data Management & System Prioritization	• test, by simulation in part of the Habitat area, shutdown procedures and emergency safing priorities • test data save priorities and retrieval on restart	• Mars mission priorities for shutdown must safe some habitable space and allow crew to fix problem • computer programs must know, & save critical data; full reprogramming may take too long	• a controlled lunar test will determine the limits of a safe shutdown and the amount of repair under time constraints that can be accomplished IVA & EVA in reduced gravity. This will aid in determining when to fix a problem at the start, when to evacuate and how to recover the systems.
NTP/Cryo	engine/propulsion system	• verify engine firing in-space (NTP) • verify function after long duration (180 - 512 days) in space (both) • retrieve propulsion system and engine for teardown, inspection and evaluation (both)	• Mars system must work for mission success & crew safe return • engine restart critical to mission success and safe crew return	• An engine that has under gone a space trial can be highly instrumented, more so than the mission vehicle. Its functioning can be examined remotely, refined & retested in space. The NTP system, after a cool down period can be retrieved for an inspection of parts. Wear on components, any deterioration due to firing or space related exposure can be found and corrected before a Mars commit
Operations	Resupply	• establish methods & techniques of manual and log module operational requirements	• methods of resupply will closely parallel the ones used on the lunar outpost.	• The endurance of people, the limitations of machinery and the viability of the techniques of manual and logistics module resupply must be identified before the Mars crew must perform these functions & explore over the 500 - 600 day period

Figure 4-8. Lunar - Mars Testing Elements (Concluded)

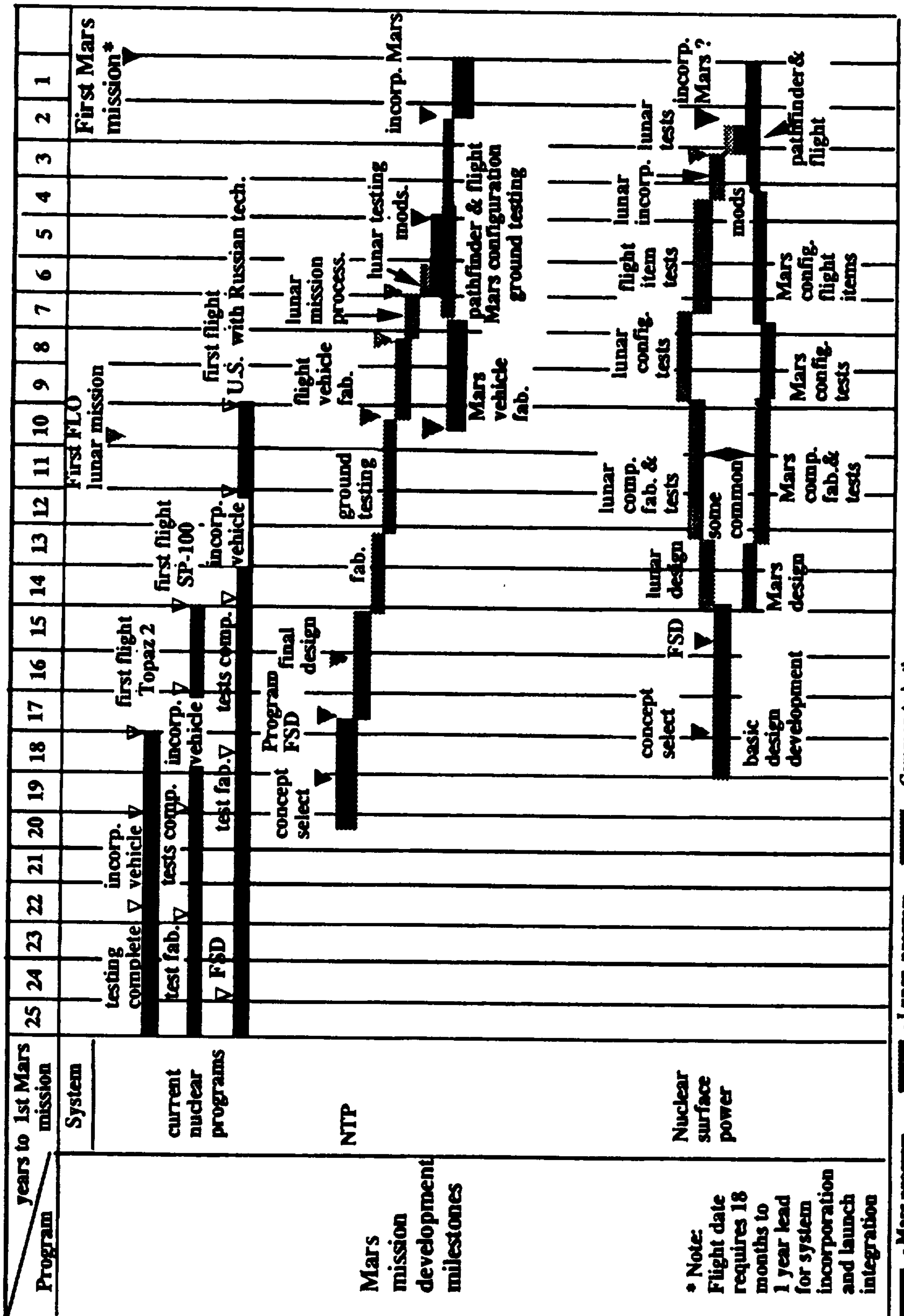

Figure 4-9. Development Schedule to First mars Flight

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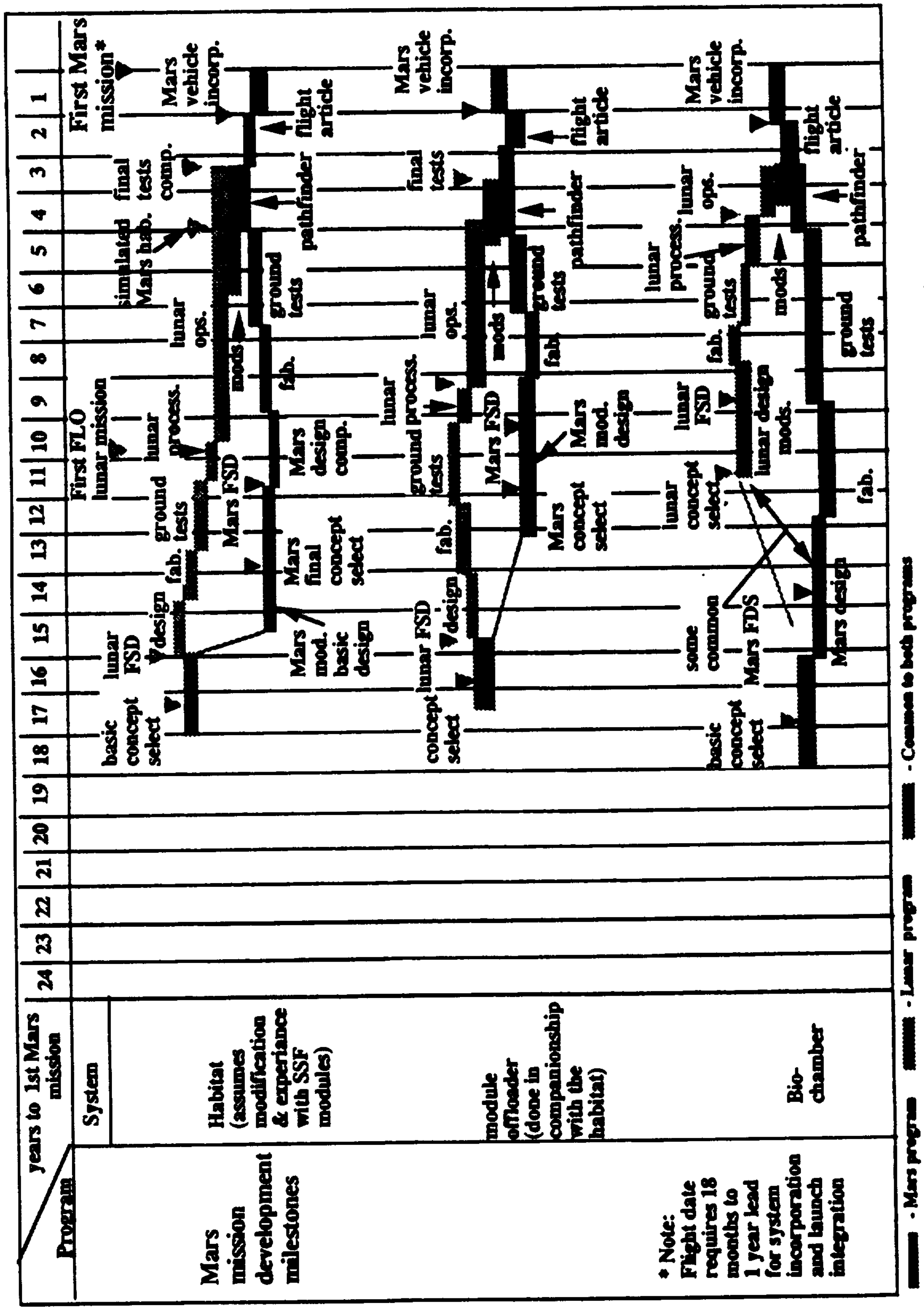

Figure 4-10. Development Scehdule to First Mars Flight - Continued

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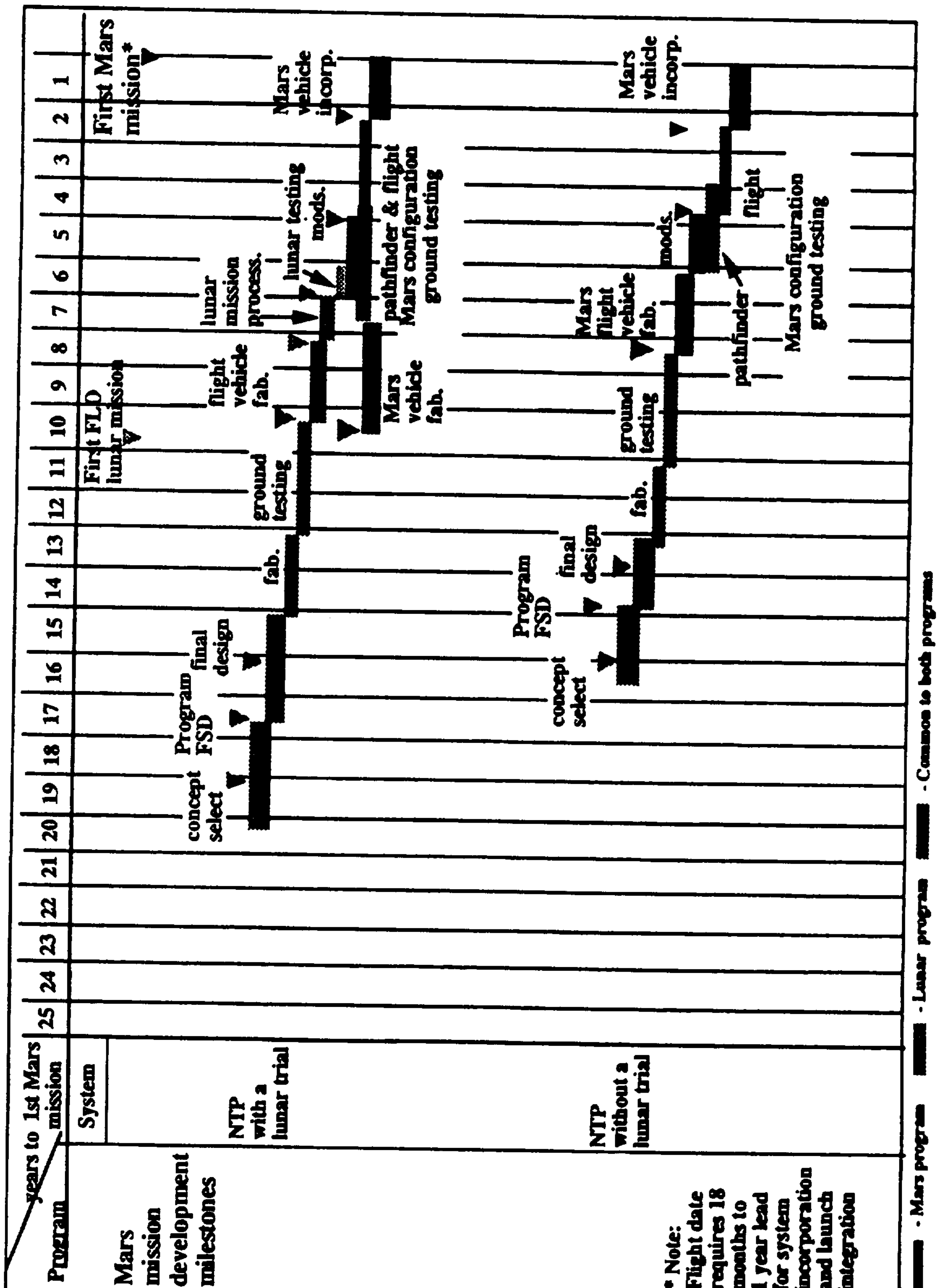

Figure 4-11. Development Schedule to First Mars Flight - NTP Comparison

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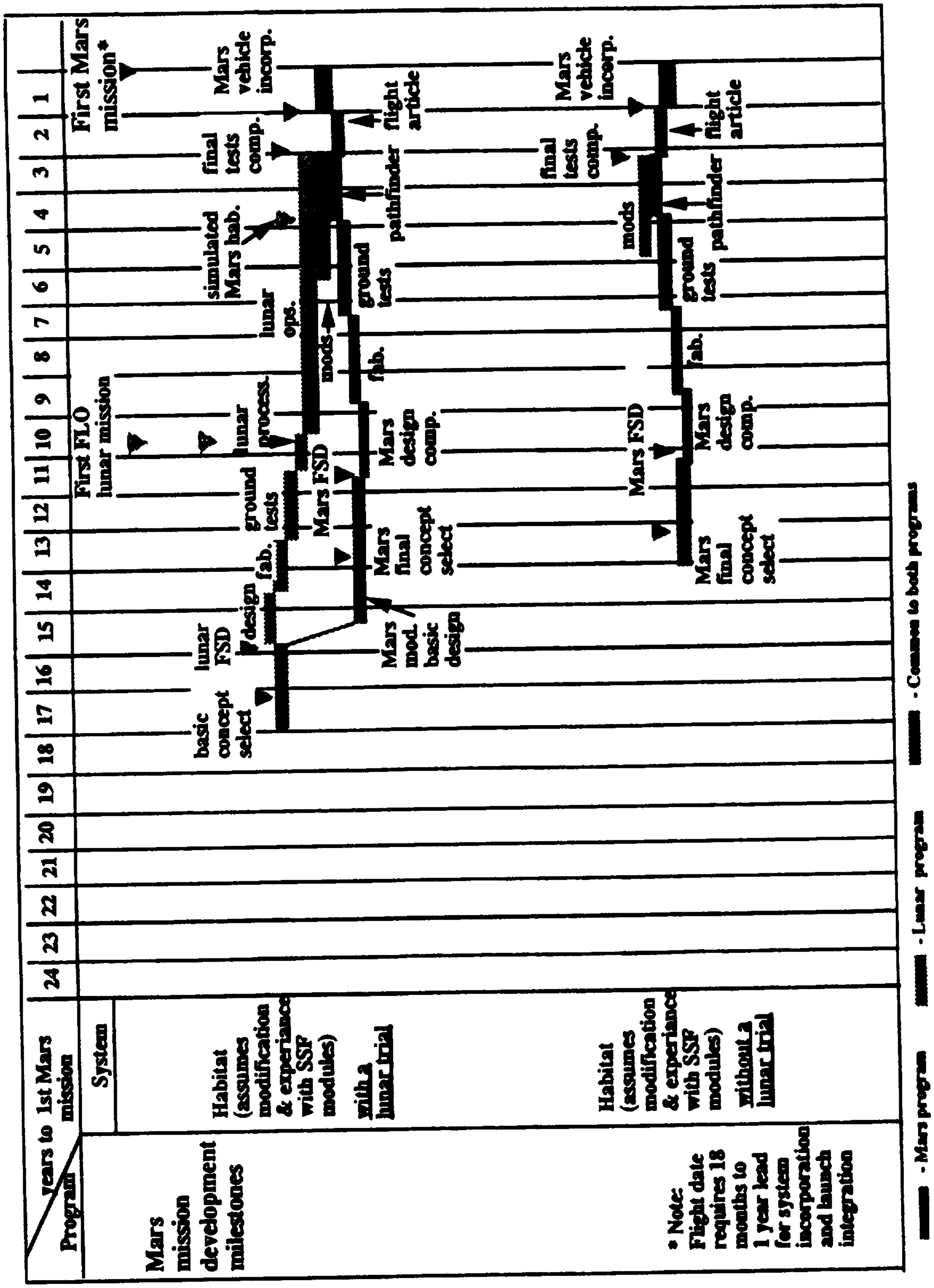

Figure 4-12. Development Schedule to First Mars Flight - Habitat Comp.

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Habitable Volume 42 m^3	(FLO Baseline - 32 m^3)
Usable Floor Ares 21 m^2	(FLO BAseline - 13 m^3)

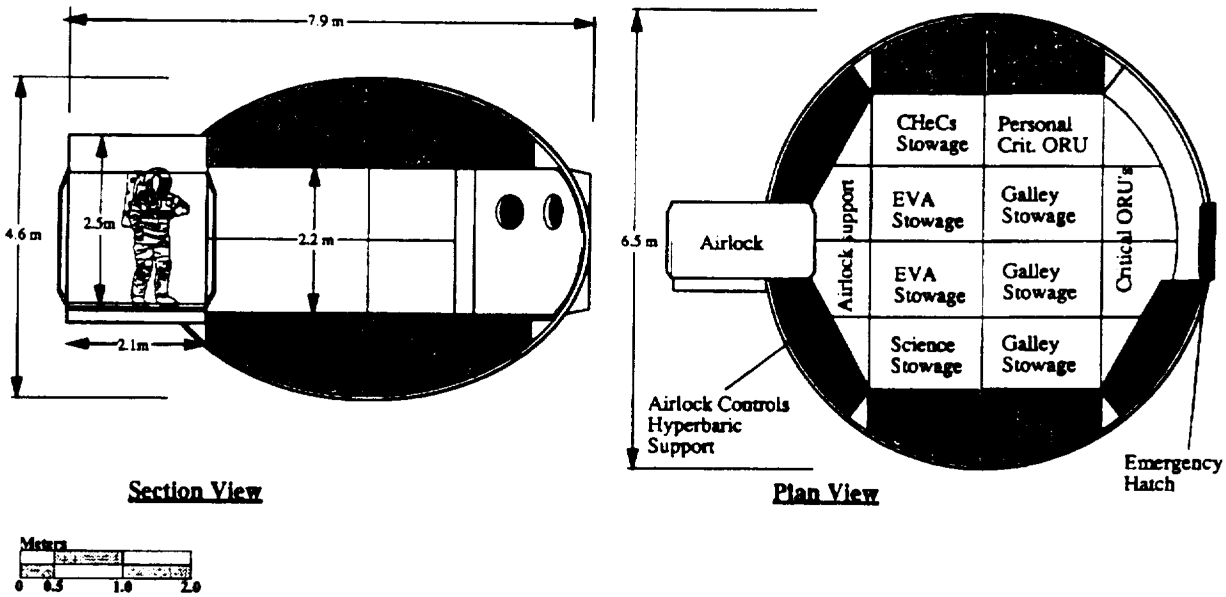

Figure 4-13. FLO Ellipsoidal Habitat Option

	Volume allocated in Baseline FLO Hab	Above Ceiling	Deck Level	Below Deck	Distributed Systems
Airlock Support	1 0	0.0	0.0	1.5	0.0
Depress Pump Assembly	1 0	1.0	0.0	0.0	1.0
SCPU's	2 0	0.0	2.2	0.0	0.0
EVA Stowage	3 0	0.0	0.0	2.0	0.0
AL controls/Hyperbaric Support	3 0	0.0	2.2	0.0	0.0
CHeCs/Stowage	3 0	0.0	2.8	0.0	0.0
Science	3 0	0.0	2.8	0.0	0.0
Science Stowage	0 0	0.0	0.0	1.2	0.0
DMS Comm.	1 0	0.0	2.2	0.0	0.0
Hygiene/WMF	2 0	0.0	2.8	0.0	0.0
Galley	2 0	0.0	2.0	0.0	0.0
Galley Stowage	4 0	0.0	0.8	4.0	0.0
Personal Stowage	1 0	0.0	0.0	2.2	0.0
Critical ORU's	2 0	0.0	0.0	2.4	0.0
OPS Stowage	1 8	0.0	0.0	2.4	0.0
ECLS	14 0	14.0	0.0	0.0	14.0
Utility "Standoff" volume	13 0	0.0	0.0	0.0	7.9
Airlock Intrusion	5 3	0.2	1.5	0.2	0.2
Rack "Swing" Space	4 8	NA	NA	NA	NA
Endcone Dist. Systems	1 4	NA	NA	NA	NA
Usable Endcone Volume	1 6	NA	NA	NA	NA
Habitable Volume	31 7	0.0	42.5	0.0	0.0
Totals	101 6	15 2	61 8	17 7	7 0 = 101.7 m^3

Figure 4-14. Ellipsoidal FLO Hab Volume Analysis

5.0 MARS RESUPPLY AND EVOLUTION

5.1 INTRODUCTION

As part of the operations analysis for a Mars evolution scenario, a preliminary analysis for resupply to a Mars mission sequence that transitions from single manned missions to permanent occupancy was generated. This satisfies the Statement of Work requirement to analyze transition to crew rotation and resupply. For this analysis, permanent occupancy was established by overlapping conjunction missions with a cargo support launch between each manned flight. An alternative scenario does exist. It is possible to employ conjunction and opposition missions alternately for building the base and establishing permanent occupancy. This would be done by using a series of conjunction missions to establish the base, then using opposition mission profiles to perform permanent base crew rotation. The crew would not spend as long on the surface as with overlapping conjunction missions (2 to 3 years versus 3 to 4 years). This second scenario was not pursued for this analyses, but should be a subject of future study.

5.2 SCENARIO DESCRIPTION

Using the conjunction mission only series, and without changing the abort-to-surface criteria, all consumables would be delivered in the cargo flights going directly to the surface. The crew flights will have transit consumables on board, but these remain on board and are not counted as part of the surface supplies. However, some consumables may be brought to the surface as part of the extended missions manifests to cycle through and augment surface supplies. The mass per day per person is estimated to be 4 kilograms (kg) divided as follows: 2.5 kg water, 0.7 kg dry food, 0.8 kg air/packaging/ other items. This number is without any mass allocation for spares, except medical and life support systems. Two lines of inquiry were followed for crew consumables supplies, one with the use of a large scale Bio-chamber to grow food in situ and one without it. As a starting safety requirement, a consumable reserve that would last for 2 to 3 years of unsupported surface staytime, not including the Bio-chamber, was to be maintained after the first manned mission in the event of a missed mission window, cargo flight failure or other mishap.

The Bio-chamber contribution to the consumables would be approximately 1.6 kg per person, of that 1.0 kg per day is water and 0.6 kg is food. This accounts for the difference between wet and dry food with some drinking water. Dry food, such as cheese, nuts, sugar and spice items, and canned goods such as meat and fish would comprise the remaining 0.1 kg of consumables. In all, this would comprise 40% of the total transported consumables mass. A better percentage could be obtained, if there was full closure on the water. For this study, this full water closure line was not done. The Bio-chamber for a crew of twelve (a permanent base with crew overlap) will have a mass between 50 and 60 tonnes; 70% of which is chamber mass with initial operating equipment and the rest being spares, stores and refurbishment items specific to the Bio-chamber. Some of these stores and the refurbishment items are not needed for the initial chamber operations, therefore, the chamber itself can be launched on a single cargo flight, with any additional service items brought on a subsequent cargo mission as base stores. The main 12 crew Bio-chamber itself is to be a deployable system that will contain 120 to 140 cubic meters of growing space.

A smaller self-contained system is included in the beginning cargo mass allotment for the first manned Mars mission, but was not considered to be a main consumable supply system. That is, for the purposes of this study, the original Bio-chamber was not counted on to supply a significant amount of food and water. It was to act as a prototype for determining the best chamber operations, for psychological relief, food variety and to investigate methods of food storage and waste disposal. Both Bio-chambers will use a variety of methods of growing various crops in order to increase crop production reliability. These will include, but not be limited to hydroponics, artificial soil, and natural soil. For this study, it was considered that there would be no significant contribution to the consumables by the large Bio-chamber until three months after its initial deployment. This would allow time to checkout the systems, plant the crops and have the first harvest series.

5.3 MARS RESUPPLY AND EVOLUTION CONSUMABLES WITH BIO-CHAMBER SUPPORT

Figure 5-1 shows the schedule for establishing a Mars base through two single 6 crew manned missions evolving to extended, overlapping missions each of 6 crew that place 12 crew at Mars continuously. Also shown on this schedule are the supporting cargo flights, both to establish the base (first three cargo flights), cargo / base resupply flights and the 12-crew Bio-chamber flight. All of these cargo flights were considered to be 50 tonnes delivered to the surface. The length of the missions, the duration of stay and transit times, and the relationship of mission opportunities is based on relative generic opportunities, with the exception of the first two missions. The first two missions are based on the current 2011 and 2014 missions, the rest are relative opportunities in which the actual dates of departure and arrival, the staytime and mission overlap are to be determined.

Each of the cargo mission has listed the amount of cargo mass (50 t.) and what portion of that is devoted to consumables. The basis for these numbers is given in figure 5-2. Note again that the required amount consumables is impacted by the large Bio-chamber only after it has been deployed three months. This figure give an accounting of the consumables and their use throughout evolution transitions. At the

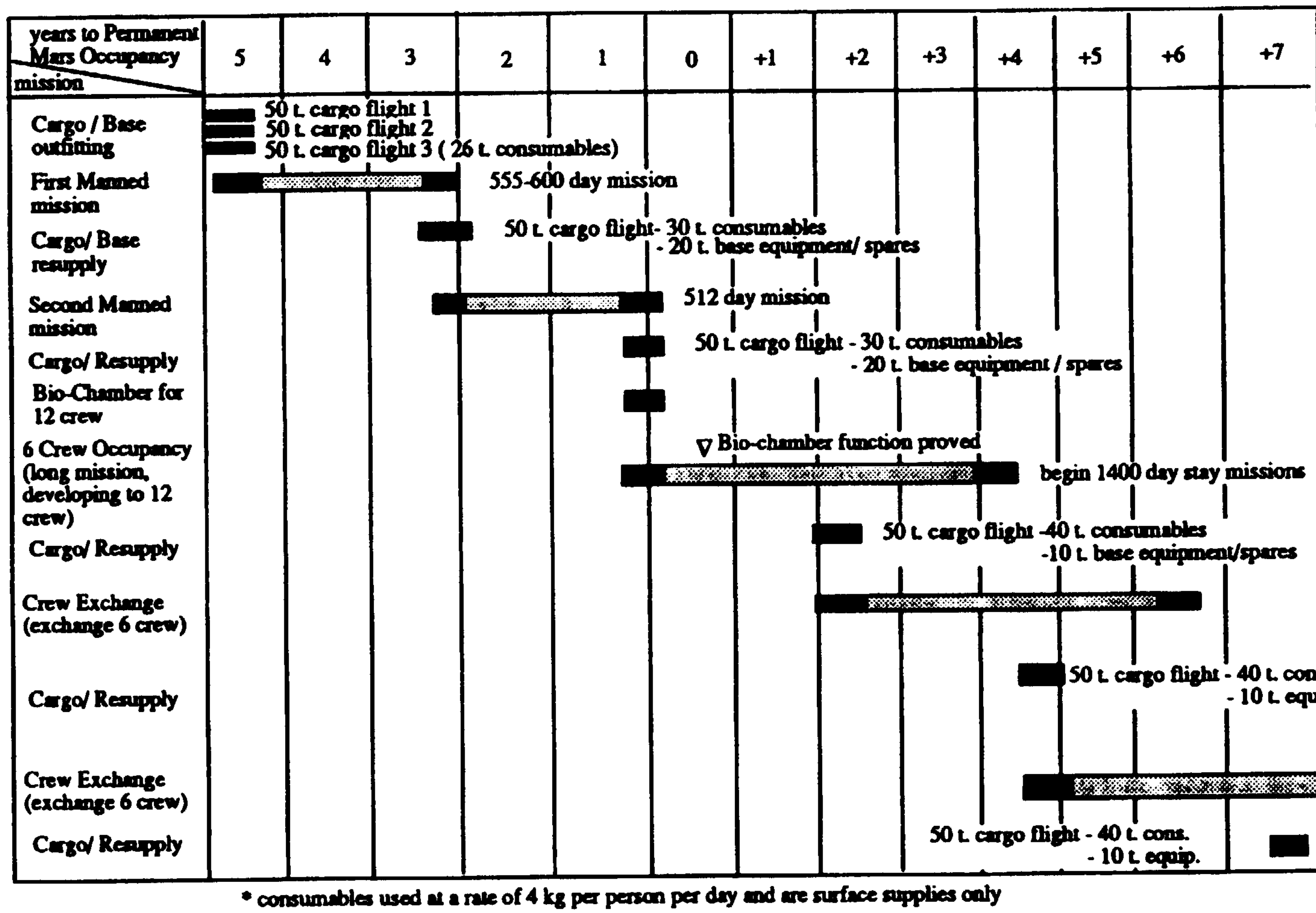

Figure 5-1. Mars Resupply and Evolution, Surface Consumables with Bio-chamber Support

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Mission	# Flights	Consumable Mass (kg) brought	Brought Consumables at Mission Start (kg)	Days supported at mission start	Consumables used	Bio-chamber contribution %total/kg	Reserve Brought Consumables (kg)	Reserve Consumables days	Crew supported
Cargo/Base outfitting	3	26,000	—	—		0	26,000	1,083	6
First Manned mission	1	—	26,000	1,083	14,400	0	11,600	483	6
Cargo/Base resupply	1	30,000	—	—	—	0	41,600	1,733	6
Second Manned mission	1	—	41,600	1,733	12,288	0	29,312	1,221	6
Cargo/Resupply	1	30,000	—	—	—	0	59,312	2,471	6
Bio-Chamber for 12 crew	1	—	—	—	—	—	—	—	—
6 crew Occupancy (long mission, developing to 12 crew, with missions overlapping)	1	—	59,312	2,471	19,200 (600 days initial mission, 200 day extension)	40% three months after start 6,816	46,928	1,955	6
Cargo/Resupply	1	40,000	86,928	1,811	28,800 (600 days)	40% 11,520	69,648	1,451	12
Crew Exchange (exchange 6 crew)	1	—	69,648	1,451	38,400	40% 15,360	46,608	971	12
Cargo/Resupply	1	40,000	86,608	1,804	28,800	40% 11,520	69,328	1,444	12
Crew Exchange (exchange 6 crew)	1	—	69,328	1,444	38,400	40% 15,360	46,288	964	12
Cargo/Resupply	1	40,000	86,288	1,798	28,800	40% 11,520	69,008	1,437	12

4 kg/day/person = 2.5 kg water
0.7 kg dry food
0.6 air/packaging/other
Bio-chamber contributions = 1.6 kg/day/person (@ 40% total)
(~1.0 kg water & 0.6 kg food)

Figure 5-2. Mars Evolution Missions Surface Consumables with Bio-Chamber Support

time the permanent occupancy and crew exchanges are established, the consumables that need to be resupplied and the bio-chamber contributions have almost leveled off. There is a slight decrease in the amount of consumables on hand as the missions progress, but not a rapid drop. The resupply requirement becomes almost stable. While an additional cargo flight may eventually be needed, it is not something that must be done this early in the program. This can be seen more clearly in the bar chart of figure 5-3 and the line chart of figure 5-4.

5.4 MARS RESUPPLY AND EVOLUTION CONSUMABLES WITHOUT BIO-CHAMBER SUPPORT

Figure 5-5 is the same flight series depicted in figure 5-1, except there is no Bio-chamber flight. The total mass allotment for each cargo flight now must be dedicated to resupply consumables for the permanent base. Even this is not enough to maintain both the permanent base and the safety reserve. Additional supplies must be brought on the manned mission flights using mass that would normally go to base equipment. The inventory numbers are shown in figure 5-6. Reviewing these numbers indicates that the

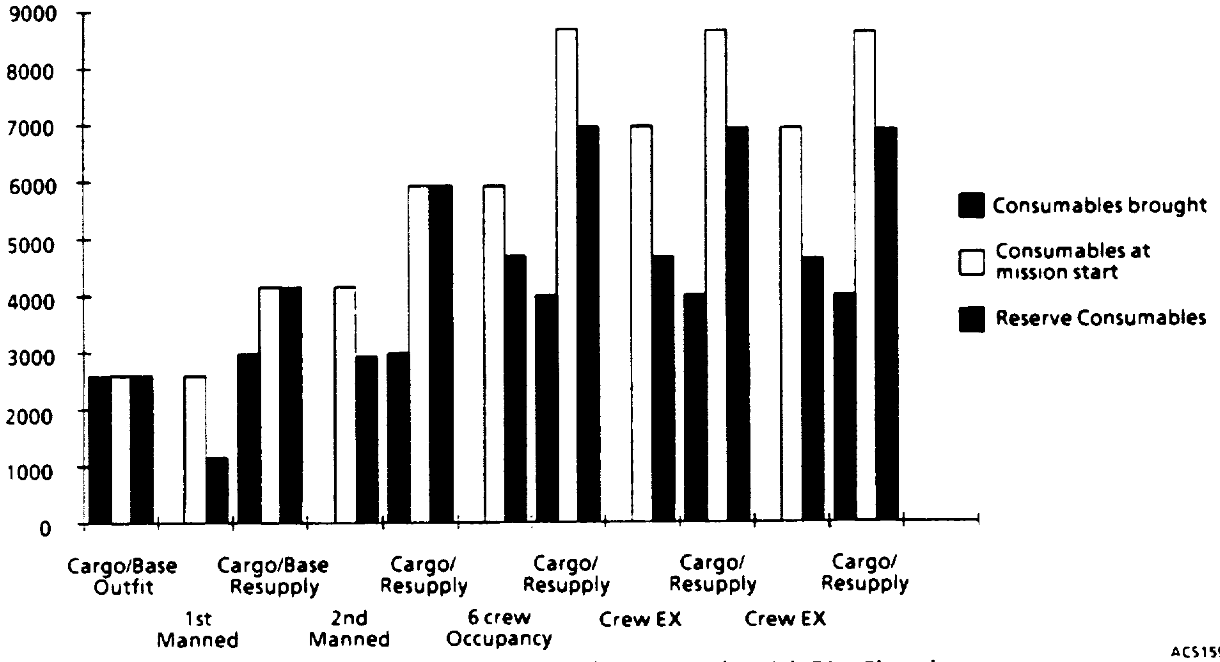

Figure 5-3. Mars Consumables Resupply with Bio-Chamber

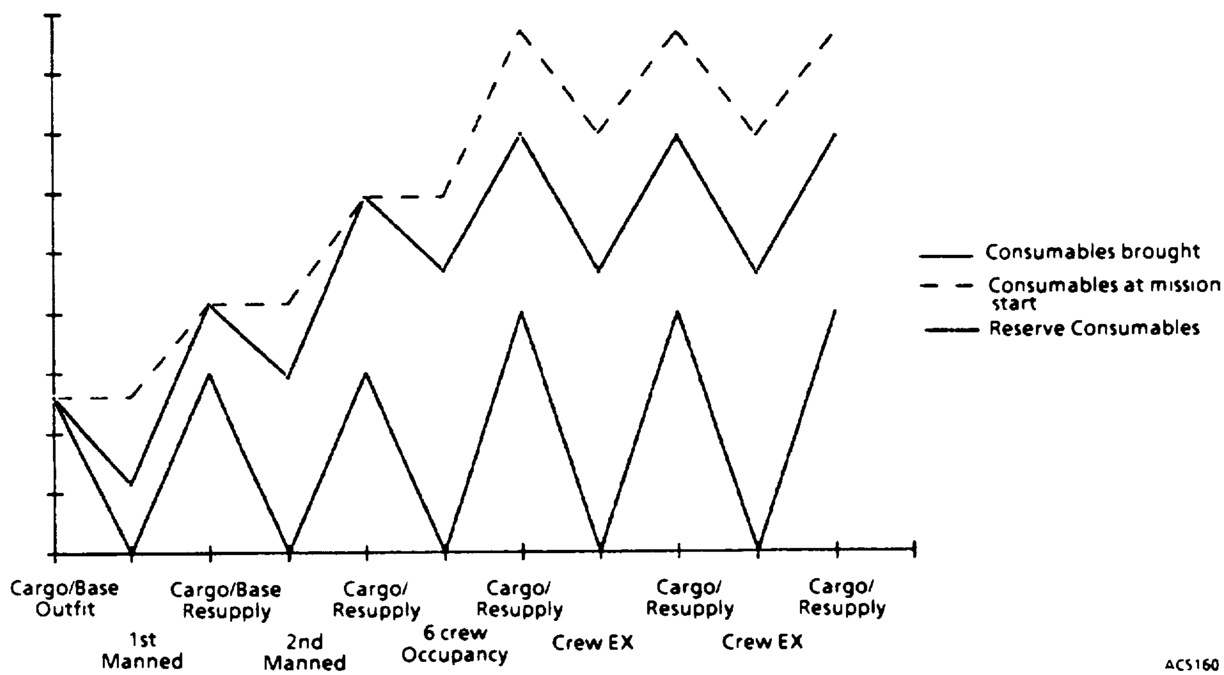

Figure 5-4. Mars Resupply without Bio-Chamber

resupply operation is not stable. Despite the fact that the initial consumable reserves are greater than with the Bio-chamber mission, the they deplete faster with this scenario. Soon after the missions depicted here an additional consumables resupply mission must be sent to make up the dwindling reserves. This can be seen in the bar chart of figure 5-7 and the line chart of figure 5-8.

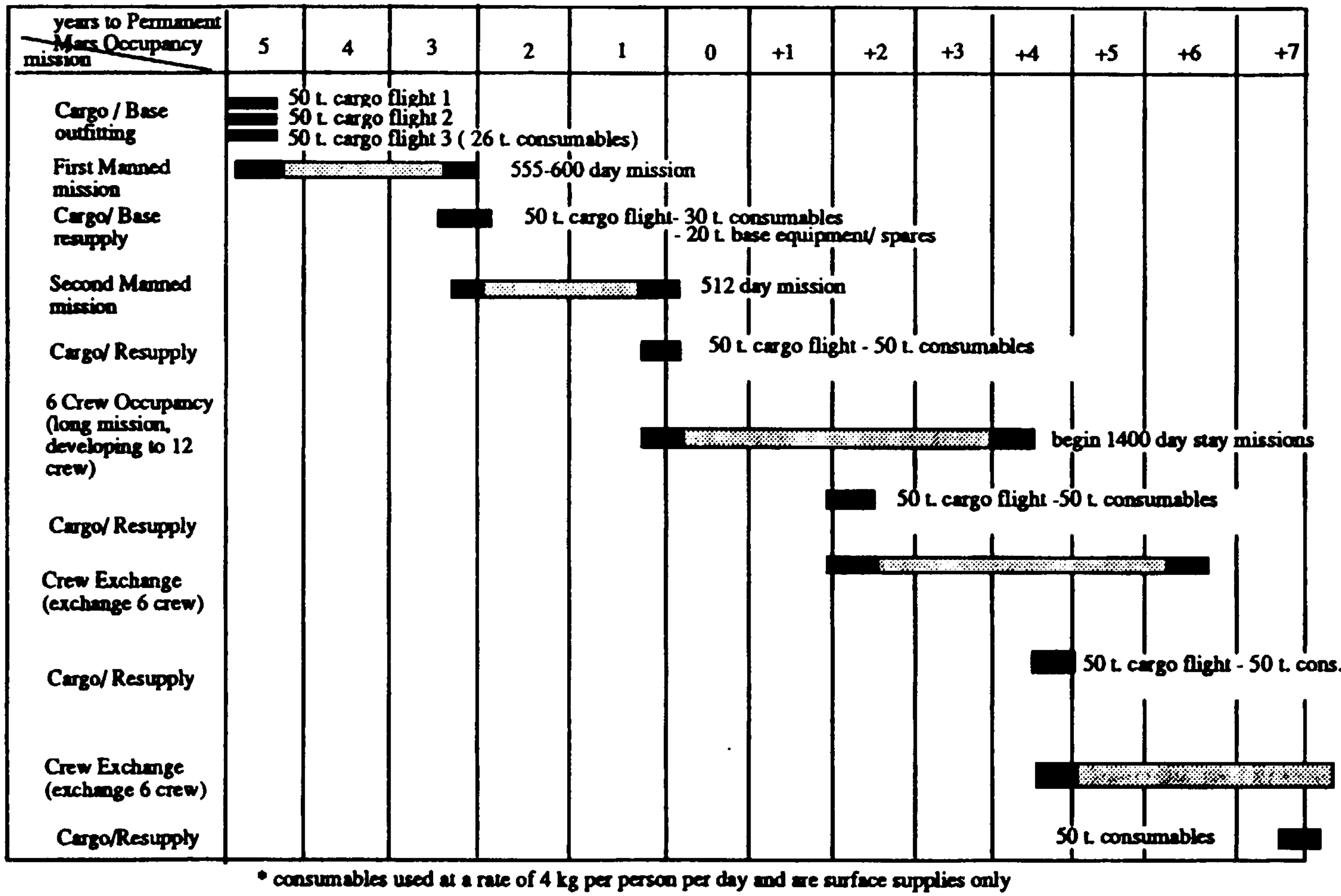

Figure 5-5. Mars Resupply and Evolution, Surface Consumables Without Bio-Chamber Support

ACS161

Mission	# Flights	Consum-able Mass (kg) brought	Brought Consum-ables at Mission Start (kg)	Days supported at mission start	Consum-ables used	Bio-chamber contribution %total/kg	Reserve Consum-ables (kg)	Reserve Consum-ables days	Crew supported
Cargo/Base outfitting	3	26,000	—	—		0	26,000	1,083	6
First Manned mission	1	—	26,000	1,083	14,400	0	11,600	483	6
Cargo/Base resupply	1	30,000	—	—	—	0	41,600	1,733	6
Second Manned mission	1	—	41,600	1,733	12,288	0	29,312	1,221	6
Cargo/Resupply	1	50,000	—	—	—	0	79,312	3,305	6
6 crew Occupancy (long mission, developing to 12 crew, with missions overlapping)	1	5,000	84,312	3,513 for 6 crew	19200 (600 days initial mission, 200 day extension)	0	65,112	2,713	6
Cargo/Resupply	1	50,000	115,112	2,398 for 12 crew	28,800 (600 days)	0	88,312	1,798	12
Crew Exchange (exchange 6 crew)	1	5,000	91,312	1,902	38,400	0	52,912	1,102	12
Cargo/Resupply	1	50,000	102,912	2,144	28,800	0	74,112	1,544	12
Crew Exchange (exchange 6 crew)	1	5,000	79,112	1,648	38,400	0	40,712	848	12
Cargo/Resupply	1	50,000	90,712	1,890	28,800	0	61,912	1,290	12

4 kg/day/person = 2 5 kg water
0 7 kg dry food
0 6 air/packaging/other

Figure 5-6. Mars Evolution Missions Surface Consumables with Bio-Chamber Support

5.5 RESULTS

This initial results indicated that the Bio-chamber will be important for permanent Mars occupancy. Without it the consumables use all the cargo flight capacity at one cargo flight per mission. This was the assigned cargo flight requirement after the initial base buildup. No additional base buildup equipment, facilities or spares can be sent under these circumstances. A true spares assessment was not done for this study and it is possible that with such an assessment that two cargo flights per manned mission may be required. It is probable that an additional spares/equipment flight will not help the consumables problem and indeed, part of the equipment flight will be devoted to consumables. The impact of the consumables, spares,base buildup equipment and facilities must be considered together to get a true picture of the Mars evolution program requirements. This includes a review of the risks and abort scenarios that will be incurred in the program scenario, including the placement of additional landers/ ascent vehicles, repair capability and on-orbit consumable caches.

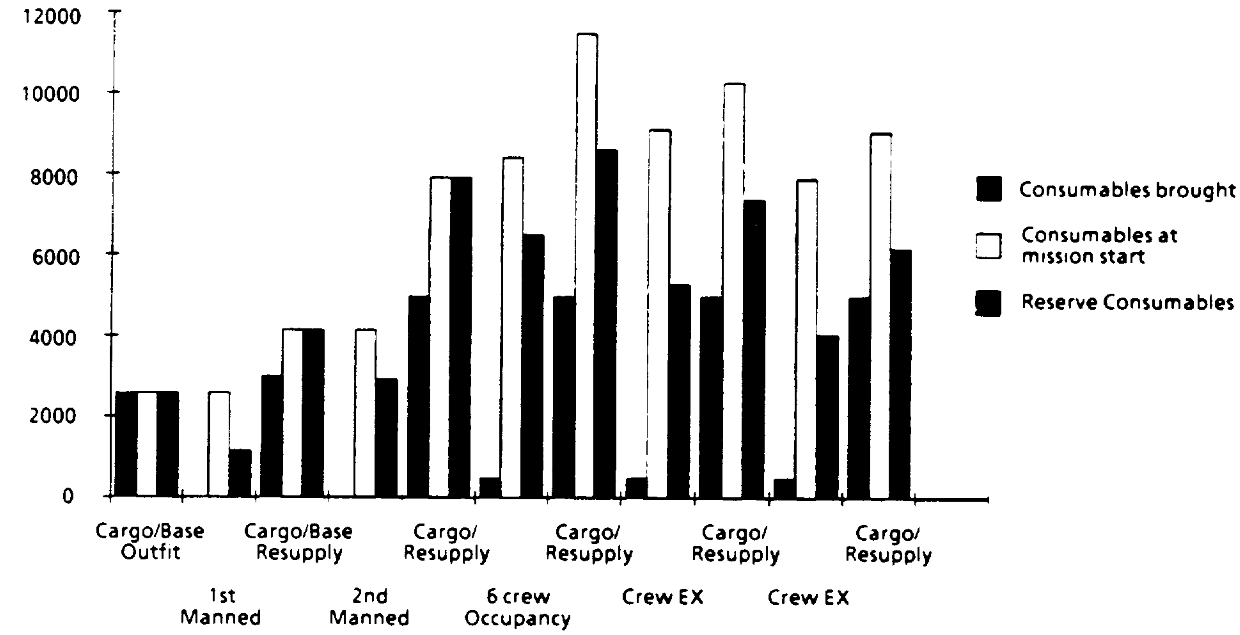

Figure 5-7. Mars Consumables Resupply Without Bio-Chamber

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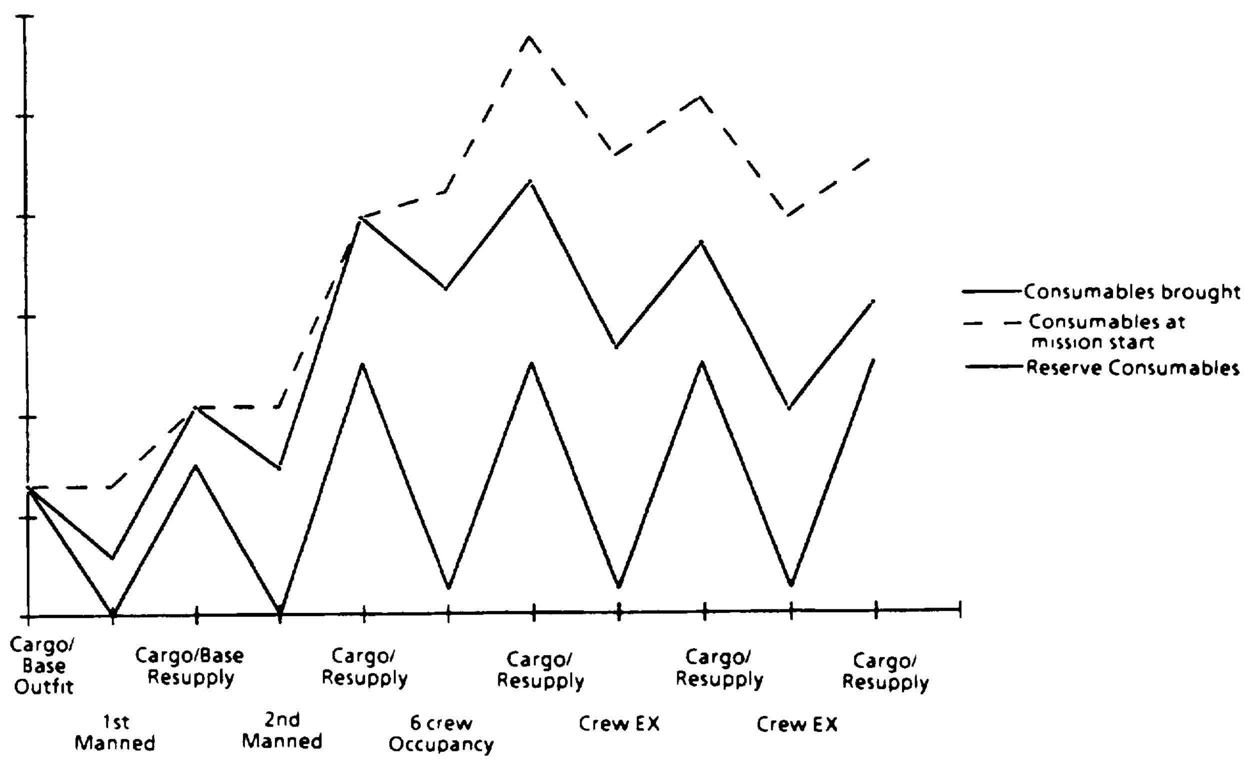

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Figure 5-8. Mars resupply Without Bio-Chamber

6.0 LOW THRUST TRAJECTORY CHARACTERISTICS

6.1 INTRODUCTION AND METHOD

This section was adapted from Boeing IR&D work aimed at developing better methods for mission profile analysis. It illuminates certain characteristics of low-thrust trajectories that are not generally appreciated. These characteristics lead to a means of approximate analysis of low-thrust profiles using delta V and trip time, without the necessity of using a low-thrust path optimizer for every mission profile to be analyzed. The approximations are about as good as those typically used for high-thrust profile analysis.

Generally, the heliocentric trajectory that is followed by a transportation vehicle utilizing low-thrust propulsion is similar to the trajectory of a vehicle which is propelled with a high-thrust system. A sample of each category of trajectory is shown in figure 6-1. Wile the low-thrust trajectory is more of a spiral, the actual shape of the path is determined by the amount of time spent burning and coasting throughout the trip. Typical optimal low-thrust paths have thrusting periods for planetary departure and arrival and a coast in between. If the path passes closer to the Sun than either planet, a third thrusting period during perihelion passage may be optimal. This is analogous to the use of a "deep-space maneuver" by a high-thrust system. As the percentage of the trip time spent burning decreases (corresponding to an increase in the percentage of the trip spent coasting) the low thrust trajectory begins to approximate a high-thrust path. This is due to the fact that the burns are becoming more nearly instantaneous, the assumption made for high-thrust trajectory analysis.

Another property of low-thrust trajectories is called the Tsien Limit. This result states that for a given opportunity of a low-thrust mission, as the trip time increase the cost (in terms of delta velocity) of the mission approaches the difference in velocity between the departing and arrival planets. The Tsien limit is strictly true only for circular, coplanar planetary paths but is a good approximation to the "very low thrust" limit for actual interplanetary trajectories since they are nearly circular and coplanar. Actual low-thrust paths between Earth and Mars approach the Tsien limit for one-way heliocentric trip times of about 500 day or more.

For shorter trips, i.e., those in the range of interest, the delta V becomes highly sensitive to trip time. As trip time is reduced, the delta V increases and the ability of the low-thrust system to deliver delta V decreases (since it can only deliver a certain amount of delta V per unit time). At some point the trends cross; the low-thrust system must thrust constantly at this point and no shorter trip time is possible.

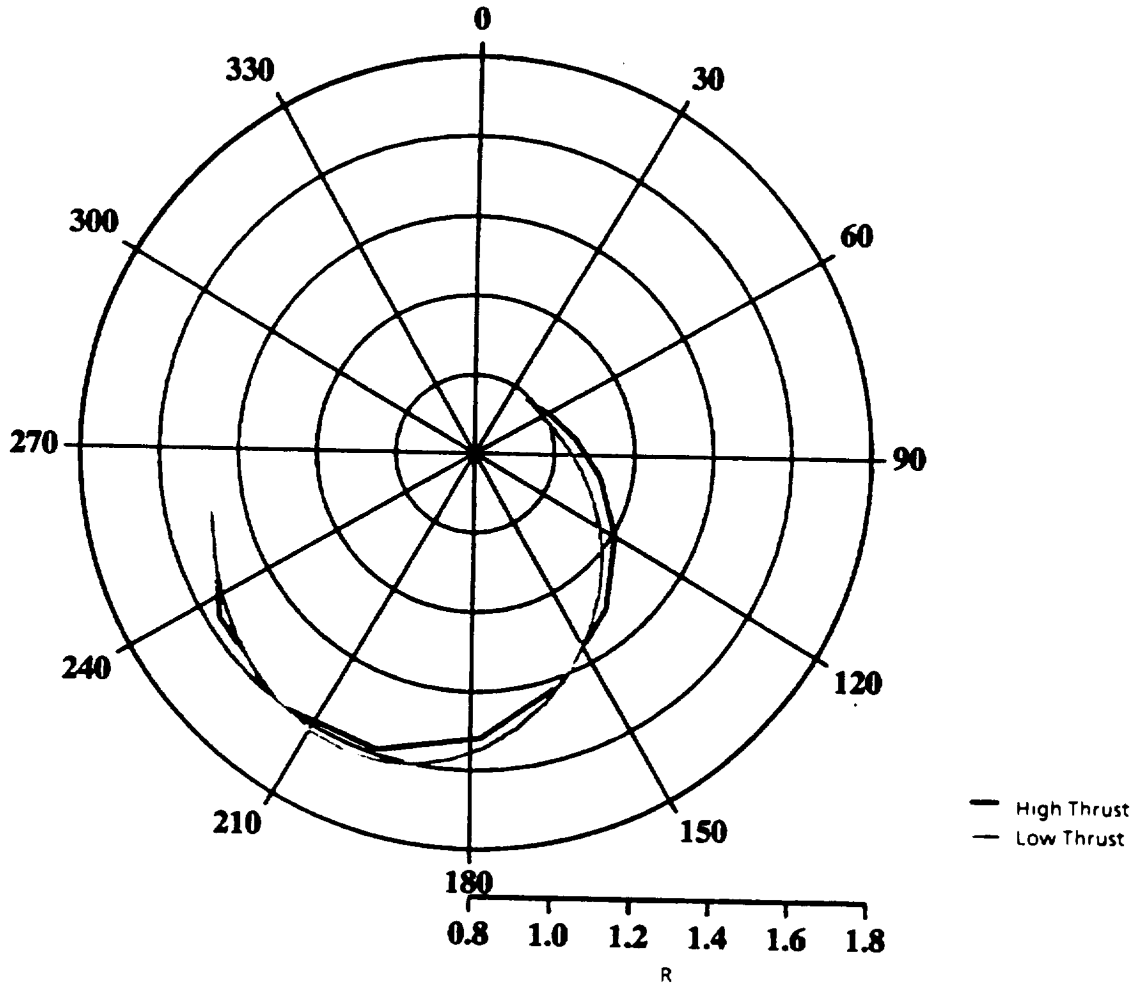

Figure 6-1. High Thrust - Low thrust Trajectory Comparison

The approach discussed here depends on isolating those characteristics of low-thrust systems that affect the delta V and ignoring second-order variances. It turns out that the important characteristics are the positions of the planets (the boundary conditions), the transfer time, and the percentage of time spent delivering thrust.

Low-thrust systems delta V, as for high-thrust systems, is relatively insensitive to Isp. As usually analyzed, low-thrust systems appear sensitive to Isp. This is because as Isp is increased (for example), the thrust available at fixed electrical power is decreased and the percentage of trip time required to deliver the necessary delta V increases. *The truly important parameter is thrusting tie fraction, not Isp.* We demonstrated that by comparing trajectories where Isp and power per unit mass were both doubled, leading to essentially constant thrusting time. The delta Vs were nearly the same.

The key to the method is the use of contour plots, a method often used in STCAEM for high-thrust trajectories. On STCAEM we have strongly advocated the use of contour plots because they reveal a great many important features of a mission profile "at a glance" which are not revealed by the usual method of calculating optimal mission profile dates and paths.

6.2 COMPARISON OF LOW-THRUST AND HIGH-THRUST CONTOUR PLOTS

While a low-thrust can, at times, approximate a high-thrust trajectory, the contour plots associated with each trajectory type are very different. A contour plot gives the launch date and trip time required to reach the destination with a certain delta velocity for a particular opportunity. A sample low-thrust contour plot is shown in figure 6-2 while figure 6-3 illustrates a typical high-thrust contour plot. Both figures represent delta velocity data for the same opportunity and represent one-way trajectories from Earth to Mars. The most notable and significant difference between the two figures is that the high-thrust contour is closed (has ridges) whereas the low-thrust contour is open (has elbows). With a high-thrust mission there is a specific window (launch window) corresponding to a certain delta velocity. In other words, for a given launch date there is a finite number of arrival dates (or trip times) which will yield a specific delta velocity.

In contrast to a high-thrust mission, the contour plots associated with low-thrust trajectories are open. This means that, if the mission trip time is not a concern, any delta velocity down to the Tsien limit can be achieved with any departure date. For low-thrust missions there is no real launch window or constraint (which is definitely not true for high-thrust missions). In general, for a given launch date, as the trip time increases the delta velocity associated with the trajectory goes down. According to the Tsien Limit (discussed in the previous section) the minimum delta velocity for a specific departure date will be the difference in velocity between the launch and arrival planet.

The great difference between low-thrust and high-thrust profiles is immediately evident from the contour plots, as are such things as the optimum departure and arrival dates, the dependence of delta V on trip time and deviation from optimal dates, and the "open" verus "closed" characteristic.

6.3 TRENDING OF DELTA VELOCITY WITH TRIP TIME AND BURN TIME FRACTION

Trends of how the delta velocity changes with trip time and burn time fraction for a given low-thrust mission opportunity can be determined using the low-thrust contour plots described in the preceding paragraphs. Information regarding delta velocity, trip time, and burn time fraction can be taken form the contour plots and crossplotted on a temporary graph (an example of this is shown in figure 6-4). A third plot is then created using the crossplot to show how the delta velocity for a given opportunity is affected by

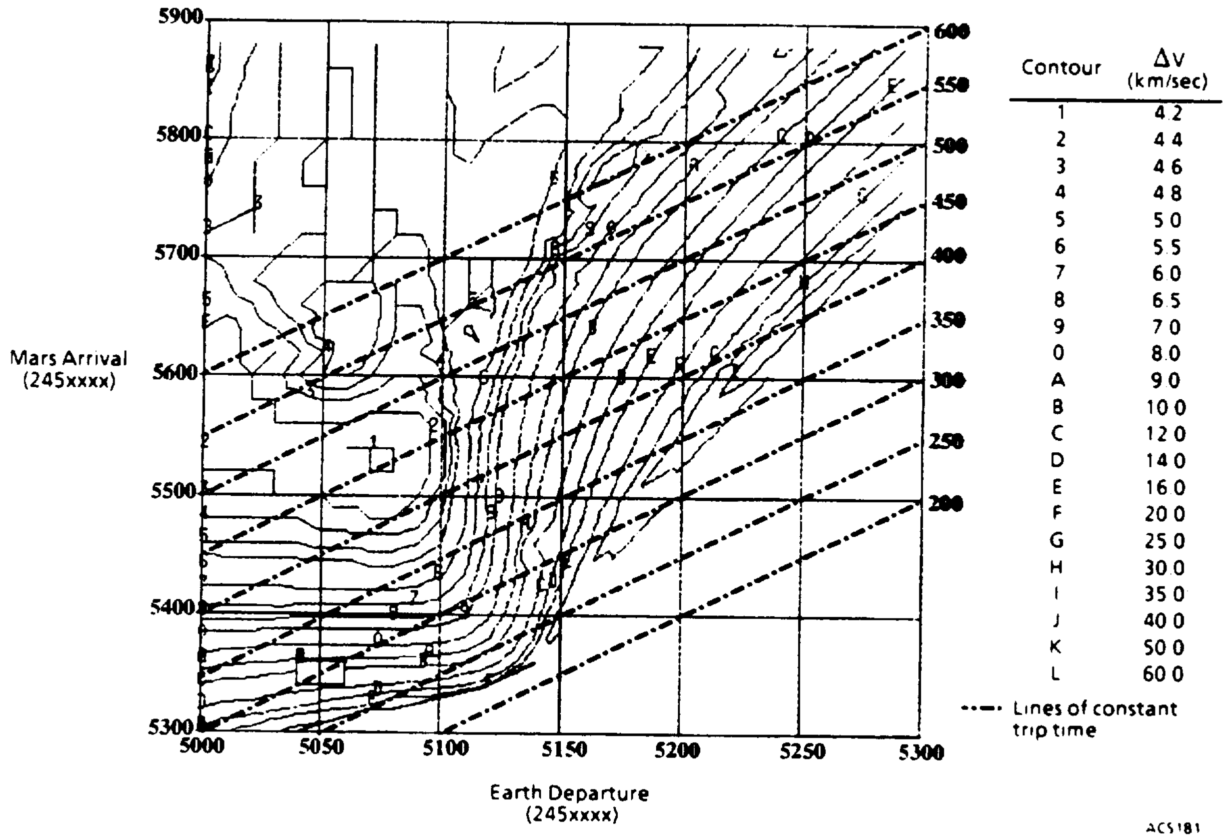

Figure 6-2. Earth Departure, Mars Arrival ΔV Contours, Isp = 7500, Power = 15 MWe, Engine Mass = 100t

changing trip times and burn time fraction. A typical sample of this plot is shown in figure 6-5. The trend that is shown in this figure is the delta velocity increases with increasing burn time fraction and decreases with increasing trip times. This result makes sense in that the larger the burn time fraction corresponds to a larger amount of propellant being used for the entire trip. Additionally, at the trip time decreases the associated delta velocity increases due to more propellant being required to make fast trips. A graph similar to figure 6-5 is show in figure 6-6 for a different low-thrust mission opportunity. It is apparent that the same trends shown in figure 6-5 can be seen in figure 6-6. The delta velocity increases with increasing burn time fraction and decreasing trip time. The sharp valleys present in figure 6-6 are due to inaccuracies in reading the initial low-thrust contour plot and the subsequent crossplots for which the data is generated by "eyeballing" the contour plots. There is every indication that if the data were more accurate, the curves shown in figure 6-6 would look very similar to those for the different opportunity shown in figure 6-5. Figure 6-7 shows a low-thrust contour plot for the subsequent Earth-Mars opportunity.

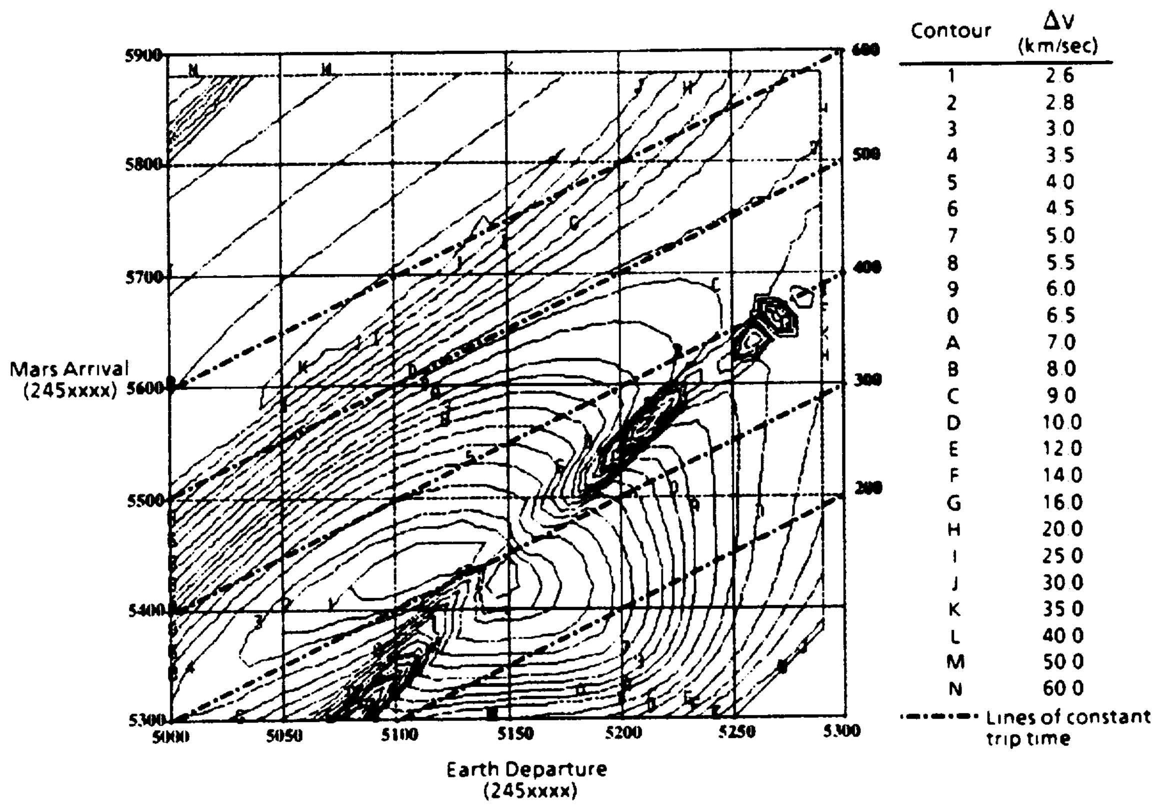

Figure 6-3. High-Thrust Trajectory Contours of ΔV

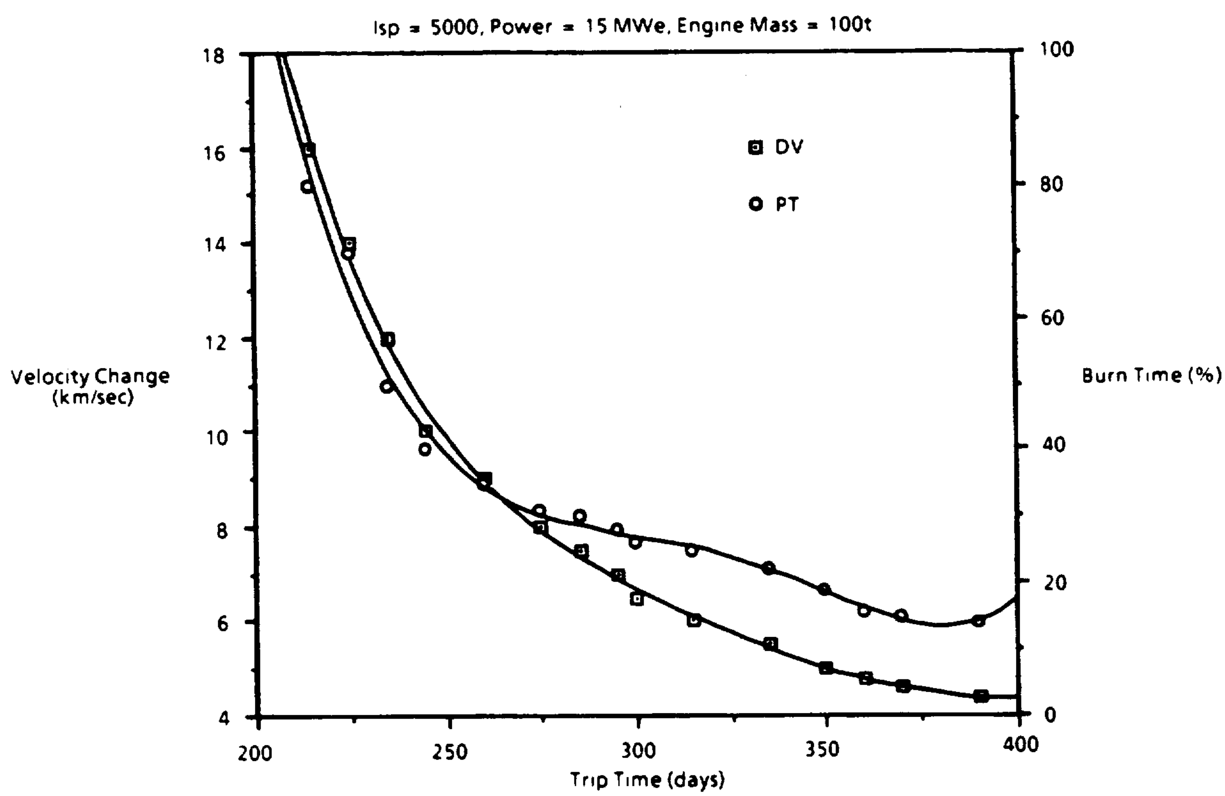

Figure 6-4. Cross Plot Derived From Contour Plot

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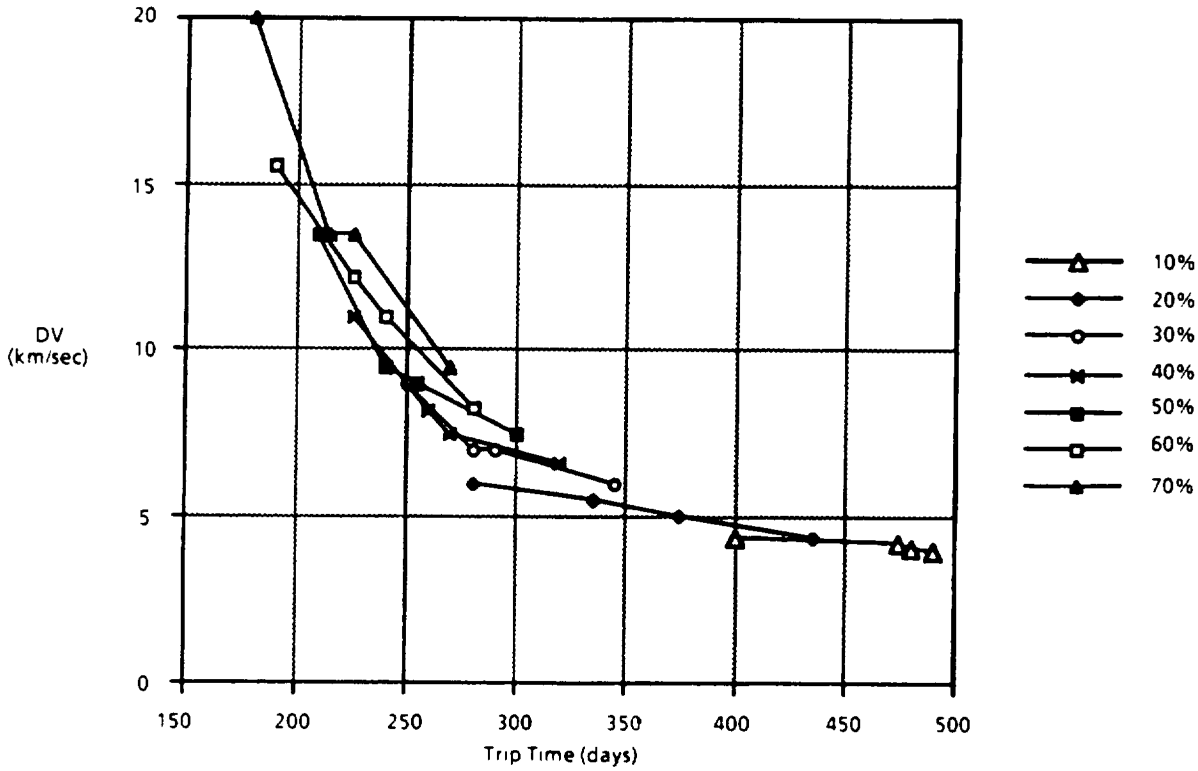

Figure 6-5. Low-Thrust Delta V versus Trip Tie and Burn Time Fraction

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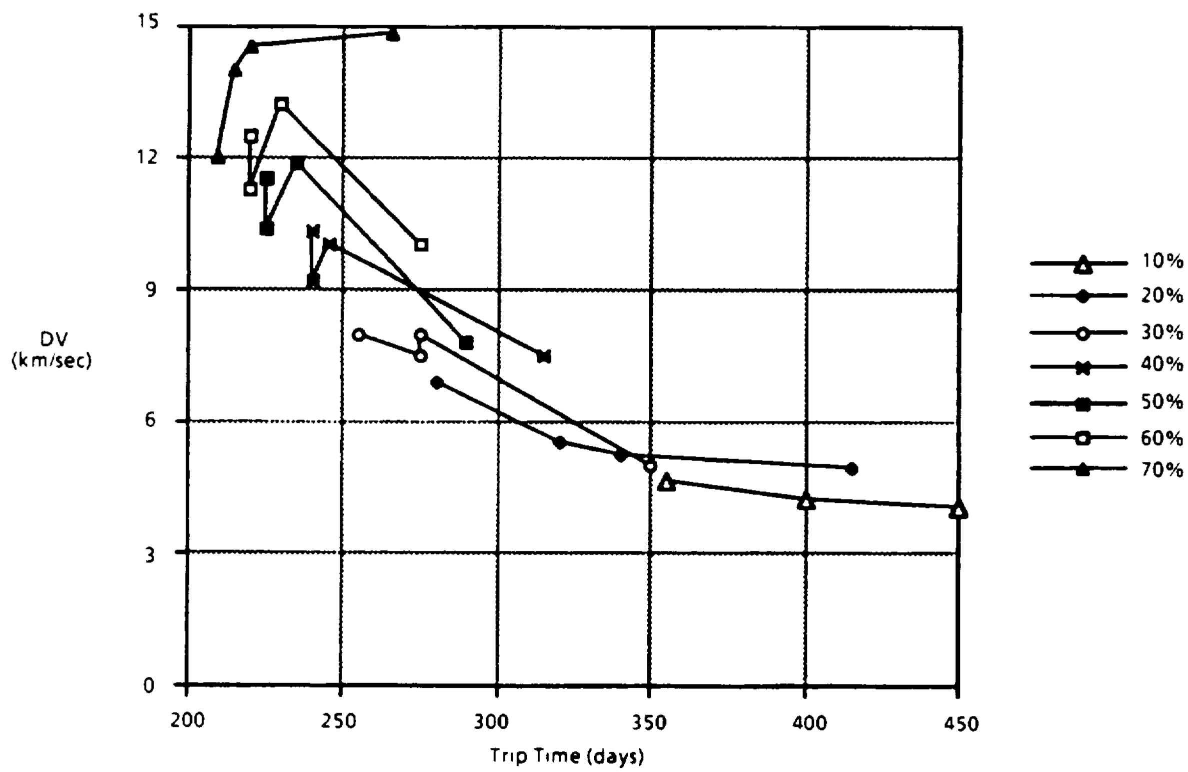

Figure 6-6. Low-Thrust Delta V versus Trip Time and Burn Time Fraction (Example 2)

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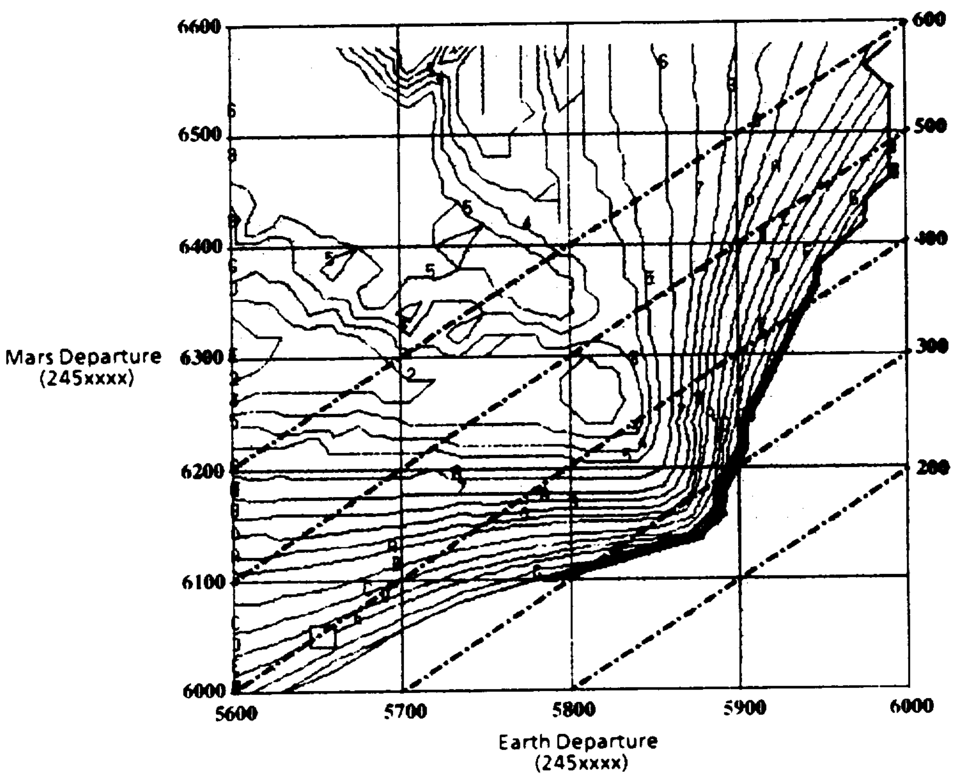

Contour	ΔV (km/sec)
1	4.2
2	4.4
3	4.6
4	4.8
5	5.0
6	5.5
7	6.0
8	6.5
9	7.0
0	7.5
A	8.0
B	9.0
C	10.0
D	11.0
E	12.0
F	14.0
G	16.0
H	18.0
I	20.0
J	25.0
K	30.0
L	35.0
M	40.0

-·-·-·· Lines of constant trip time

Figure 6-7. Earth Departure, Mars Arrival ΔV Contours, Isp = 7500, Power = 15MWe, Engine Mass = 100t

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